JPL D-12441

FLIGHT PERFORMANCE CHARACTERIZATION (FPC)

NASA Unmanned Spacecraft

Instrument Performance Report #2

Prepared by the Reliability Engineering Section of the Office of Engineering and Mission Assurance

Kin F. Man
Charles C. Gonzalez
Christine Farguson

January 31, 1995

Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California 91109

PREFACE

This is the second FPC task report describing a methodology for determining the performance of spacecraft and instrument missions. The first report provided a test of the methodology by tracking the performance of four JPL flight projects. This report adds the results two additional JPL projects, TOPEX/POSEIDON and Mars Observer.

The goal of the investigations is to relate in-flight spacecraft and instrument performance to certain causal effects, such as cost, complexity, product assurance program, and so forth. This would allow future flight projects to do risk/cost/benefit trades-offs that will be especially important to the new class of low-cost missions and smaller spacecraft.

As noted previously, four projects did not constitute a sufficient statistical base to correlate performance and the other factors; the addition of two more has not changed that significantly. Therefore detailed trends and correlations are not included in this report and they are addressed only where appropriate.

Goddard Space Flight Center has also participated in this RTOP, and has issued a report on the performance of two selected projects (COBE and UARS) using the same methodology. They are currently tracking three additional projects (EUVE, SAMPEX, and ERBS) and will issue an additional report.

ABBREVIATIONS AND ACRONYMS

AACS	Attitude and Articulation Subsystem
ADCS	Attitude and Data Control Subsystem
ALT	NASA Altimeter
ANS	Array Normal Spin
APL	Applied Physics Laboratory
ASTRA	Advanced Star Tracker
BPE	Bipropellant Equipment
BRE	Bipropellant Rocket Engine
CDH	Command and Data Handling
CDHS	Command and Data Handling Subsystem
CIS	Russian Commonwealth of Independent States
CMU	Core Memory Unit
CNES	Centre National d'ƒtudes Spatiales (French space agency)
COBE	Cosmic Background Explorer
CPM	Central Processor Modules
DFSS	Digital Fine Sun Sensor
DMSP	Defense Mapping Satellites Program
DOF	Degree of Freedom
DORIS	Determination of Orbit Radiopositioning Integrated from Satellite
DSN	Deep Space Network
EPS	Electrical Power System
ER	Electron Reflectometer
ERBS	Earth Radiation Budget Satellite
ESA	Earth Sensor Assembly
ESD	Electrostatic Discharge
EU	Expander Unit
EUVE	Extreme Ultra-violet Explorer

FPC	Flight Performance Characterization
GDS	Ground Data System
GPS	Global Positioning System
GPSDR	Global Positioning System Demonstration Receiver
GRS	Gamma Ray Spectrometer
GSFC	Goddard Space Flight Center
GSTDN	Ground Spacecraft Tracking Data Network
HGA	High Gain Antenna
HW	Hardware
IOM	Inter-Office Memorandum
IR	Infra-red
IRU	Inertial Reference Unit
ISA	Incidents/Surprises/Anomalies Reports
JPL	Jet Propulsion Laboratory
KABLE	Ka-Band Link Experiment
LGA	Low Gain Antenna
LRA	Laser Retroreflector Assembly
MAG/ER	Magnetometer/Electron Reflectometer
MMH	Monomethyl-Hydrazine
MMS	Multimission Modular Spacecraft
MO	Mars Observer
MOC	Mars Observer Camera
MOI	Mars Orbit Insertion
MOLA	Mars Observer Laser Altimeter
MOSO	Multimission Operations Systems Office
MPE	Monopropellant Equipment
NASA	National Aeronautics and Space Administration
NASCOM	NASA Communications Network
NOAA	National Oceanographic and Atmospheric Administration
NSI	NASA Standard Initiator

NTO	Nitrogen Tetroxide
OBC	On-Board Computer
OER	Office of Engineering and Review
PAPA	Product Assurance Program Assessment
PDS	Payload Data Subsystems
PFO	Problem/Failure Operations
PFR	Problem/Failure Report
PMIRR	Pressure Modulator Infrared Radiometer
POCC	Project Operations Control Center
REA	Rocket Engine Assembly
RF	Radio Frequency
RIU	Remote Interface Unit
RS	Radio Science
RTC	Real Time Commands
RTOP	Research and Technology Operation Plan
SAMPEX	olar, Anomalous, and Magnetospheric Particle Explorer
SC	Spacecraft
SEU	Single Event Upset
SSALT	Solid-State Altimeter
SSC	Stored Sequence Commands
STAREX	Star Processing Executive Program
STDN	Space-flight Tracking and Data Network
SW	Software
TBD	To Be Determined
TCM	Trajectory Correction Maneuver
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TEC	Thermoelectric Cooler
TES	Thermal Emission Spectrometer
TIROS	GSFC Meteorological Satellite (predecessor to NOAA Series)

TMON	Telemetry Monitor
TMR	TOPEX Microwave Radiometer
TOPEX	Topography Experiment
TOS	Transfer Orbit Stage
TR	Tape Recorder
TWTA	Traveling Wave Tube Amplifier
UARS	Upper Atmosphere Research Satellite
VLBI	Very Long Baseline Interferometry
XMTR	Transmitter
WFPC1	Wide Field Planetary Camera -1

SUMMARY

The Flight Performance Characterization (FPC) RTOP has developed a method for relating the performance of spacecraft (S/C) and instruments to the accomplishment of mission objectives. The goal of this task is to relate performance to certain causal effects, such as cost, complexity, and others. The results of this RTOP will provide a feedback mechanism to assist with the tailoring of product assurance requirements so that intelligent risk/cost/benefit type of trades can be made.

This report is the second deliverable of the FPC RTOP. Covered here are two NASA/JPL projects: TOPEX/POSEIDON and Mars Observer, while the first report[1] covered four projects: Voyager, Galileo, Magellan, and Wide Field Planetary Camera 1 (WFPC1). Functional performance and accomplishment of mission objectives were investigated for each project. This report seeks to provide an understanding of how the functional performance affected the mission success of TOPEX/POSEIDON and Mars Observer.

The mission objective of TOPEX/POSEIDON is primarily to study the Earth's oceans by observing the global ocean circulation. It is a joint mission between NASA and the French Space Agency (CNES), and is part of NASA's Mission to Planet Earth. This report covers 2.3 years of a three year primary and three year extended mission. So far there have been no anomalies that have prevented the S/C from fully meeting its science objectives.

Mars Observer was a NASA global mapping mission to study the surface, atmosphere, interior, and magnetic fields of Mars. After a successful, eleven month cruise to Mars, contact was lost with the S/C at the start of pressurization of the propulsion system prior to Mars orbit insertion (MOI). Nevertheless, the gravity waves and KABLE experiments were successfully completed. At the time of loss of contact, the only S/C performance deviation was the magnetic interference from the solar array circuitry that impacted the magnetometer. It was expected that measurements could still be made during solar occultation.

ACKNOWLEDGMENTS

The work described in this report was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under contract with the National Aeronautics and Space Administration (NASA). It was funded by NASA Code QT under the Research Technology Operation Plan (RTOP), UPN 623-63-03, to develop a metric for evaluating spacecraft flight performance in terms of the accomplishment of mission objectives. The opportunity provided by Code Q to develop a methodology that is useful both to NASA and the aerospace community is greatly appreciated.

The following persons have provided assistance for this work. Their efforts are also greatly appreciated:

Steven Cornford
Glenn E. Cunningham
Tom Gindorf
Mark Fujishin
Stuart Heller
Donald Lewis
Frank Palluconi
Mona Witkowski
Rod Zieger

CONTENTS

LIST OF TABLES

LIST OF FIGURES

INTRODUCTION

The objective of the Flight Performance Characterization RTOP is to track and document the in-flight performance of selected NASA missions and to provide a consistent methodology for assessing mission performance. The ultimate purpose of the FPC RTOP is to provide a feedback loop in order to assist with future design trades and product assurance program tailoring.

FPC is one of six RTOPs that fall under the general category of the PAPA (Product Assurance Program Assessment) RTOPs. The other RTOPs in this group are: Flight Anomaly Characterization, Environmental Test Effectiveness Analysis, Technical Risk Assessment, Weibul Adjusted Probability of Survival, and Product Assurance Correlation Analysis. The latter serves to tie the other five together in order to provide assessment and tailoring of product assurance practices. See Figure 1 for a general schematic of the PAPA information flow *(not provided in this version of this report*).

This report marks the second deliverable of the FPC RTOP. Covered here are two NASA/JPL projects: TOPEX/POSEIDON and Mars Observer. The first report[1] covered Voyager, Galileo, Magellan, and WFPC1. This report is broken into three parts. Part I describes the overall methodology formulated for this RTOP. Part II compares the characteristics and performance results of the S/C and instruments tracked. Finally, Part III contains the individual mission reports which describe in more detail the objectives and performance of each of the missions tracked.

Part I: Methodology

1.1 Performance

As described in the first FPC RTOP report[1], performance may have different meanings, depending on whose view point is being considered: the principal investigators, the S/C hardware engineers, system engineers, and program management. The actual measure of mission success will be perceived differently by any one of these individuals; at the same time some may view the performance as less than successful, while others view it as completely successful.

This task tracks two types of performance: *functional performance* and accomplishment of *mission objectives* (see Figure 2). *(Figure 2 is not provided in this version of this report*). Functional performance is evaluated for two types of systems: S/C systems and individual instruments. Even though recognized as significant contributors to mission success, the tracking and data, missions operations, and launch vehicle systems are not considered here in detail due to limitations in scope. However, when they impact S/C performance or mission success, the completion of mission or science objectives are also evaluated, but independently from the functional performance. Figure 3 shows the FPC process that is used to determine performance.

1.1.1 Functional Performance

Spacecraft Performance: Key Functions

For a spacecraft (S/C), the performance of each subsystem's key output functions is evaluated. To provide a sense of how well the subsystems supported the S/C system as described in Reference 1, a set of generic S/C key functions was generated for and then tailored to an individual S/C by adding or deleting as applicable. (See Table 8 for the generic list).

The S/C subsystem performance is rated for each key function, and the usage of redundancy and significant work-arounds is documented. Performance is rated through both the primary and extended missions on a scale of 1 to 5 as follows: 1= performance was met without any significant anomalies; 2= performance was met before and after a recoverable loss or temporary malfunction; 3= minor degradation in performance; 4= significant degradation in performance; 5= performance was not met. When an anomaly occurred, the type of work-around(s) employed is documented as: "R" = redundancy, "O"= operational and/or software, "M"= resource margin, and "H"= self healing. The mission impact of any non-performance is also documented as: *none, negligible, minor, moderate, significant, and severe.* And, finally, the specific impact of non-performance on individual investigations is shown. The functional performance for TOPEX/POSEIDON and Mars Observer is provided in Part III.

Overall Spacecraft Performance Rating

After each subsystem is rated, an "Overall Spacecraft Performance Rating" which measures the S/C's overall ability to support its mission is generated for each S/C. This rating takes into account both the key functional performance ratings and the mission impact of any non-performance. Therefore, performance of any non-critical functions is screened out when there is no impact on the overall mission. The same performance rating scale is also used for instruments. (It is important to note that ground support performance is not included in this assessment.) A S/C is rated with and without its payload, and through both its primary and extended (if applicable) missions. The number of redundant units that were needed to keep the mission going is also shown. The following rating scale is used: 1) S/C fully supports the mission; 2) S/C capability to support the mission is slightly diminished; 3) S/C capability to support the mission is moderately diminished; 4) S/C capability to support the mission is significantly diminished; and, 5) S/C failure. The results obtained when applying this methodology to TOPEX/POSEIDON and Mars Observer are provided in Part III.

Tracking Functional Performance

The same methodology has been used for tracking the functional performance of both S/C and instruments as described in Reference 1. This includes reviews of available flight performance reports, screening of the in-flight anomaly data, and conducting interviews with key people. The actual flight performance is determined by integrating the data and information obtained from the sources cited above and then having project personnel review the report generated. In this way, both the appropriate performance success and failure to support the mission objectives are accurately covered and verified.

1.1.2 Mission Objectives

The methodology for tracking the completion of mission or individual investigation objectives involves conducting interviews with key project personnel, namely the principal investigators and/or project scientists. This approach provides a qualitative assessment of the completion of mission objectives. It was determined that providing a quantitative measure of success of objectives completed (e.g. a percentage of objectives completed) was not always practical. For instance, measuring objectives by the quantity of data returned could be meaningless: the investigation may require only a portion of the data returned to accomplish its goals, or the unexpected availability of complimentary data may have allowed additional objectives to be met. The ability to quantify the percentage of objectives accomplished also depends on the manner in which they were originally written.

The mission objectives of TOPEX/POSEIDON were defined very specifically with a clearly stated accuracy in the sea level measurements. In contrast, the objectives of Mars Observer were stated more generally, in terms of studying the compositions, topography, and magnetic field of the planet, and the properties of its atmosphere. Therefore, for this initial effort, the objectives are evaluated in a qualitative sense (total, partial, or not met) along with a written explanation. A percentage is given only if practical to do so.

1.2 Conclusions

This report documents the extension of the methodology for determining S/C performance to two additional JPL S/C, TOPEX/POSEIDON and Mars Observer.

Recommendation:

As stated in the first report[1] a consistent set of criteria for mission operations reporting is needed to increase the efficiency and accuracy of performance reporting. In addition, if the meeting of mission objective is to be assessed quantitatively, they need to be written in such a way that allows them to be measured.

Part II: Summary of Results

2.1 Spacecraft Performance Comparisons

2.1.1 Comparison by Subsystem

Table 8 shows S/C performance comparison by key function. This table describes which subsystems have had flawless performance, and which have required various work-arounds in order to meet performance. For the following discussion, also refer to the matrices in the individual mission reports in Part III.

Telecommunications

Except for NASA communications network (NASCOM) data line problems that resulted in occasional non-recoverable real time telemetry data losses, the TOPEX/POSEIDON telecommunications performance requirements have generally been met. The only hardware problem was a momentary loss of lock of the XMTR-A downlink when XMTR-B was turned on. The latter is only used now for emergency or DSN passes.

The only performance deviation occurring with Mars Observer in the telecommunications subsystem was an antenna misalignment in its stowed position that would not impact the mapping mission.

Command and Data Handling

The TOPEX/POSEIDON Command and Data Handling Subsystem (CDHS) experienced two hardware problems on both sides of the remote interface units, RIU-6A and RIU-6B. All of the thirteen channels of the former failed due to an electrostatic discharge (ESD). Most of the channels carried engineering thermal measurements and one reported the HGA boom position. RIU-6B also had an ESD failure to the expander unit. The basic RIU-6B command interface and all other telemetry points were functioning properly, and therefore, it was decided to remain configured on RIU-6B indefinitely. The RIU hardware failures did not affect real-time telemetry significantly.

In addition to the hardware problems, there were minor software and operational procedures problems with the CDHS.

There were no hardware problems with the Mars Observer CDHS; some data losses occurred due to the ground hardware as noted in Part III.

Attitude and Articulation Control

The TOPEX/POSEIDON Attitude and Articulation Control Subsystem (AACS) had one hardware failure with the advanced star tracker, ASTRA1-B, due to a perceived high background count. The current attitude knowledge is being maintained by ASTRA1-A and the digital fine sensor. The AACS also had minor software problems that did not cause any significant degradation. There is also a minor problem with gyro jitter caused by solar array movement.

The performance of the Mars Observer AACS hardware proceeded with only temporary malfunctions and no degradation. There were some problems with the flight software, as discussed in Part III, causing a number of losses of inertial reference.

Power

The TOPEX/POSEIDON electric power subsystem met all its power requirements with no significant problems.

There were also no significant problems with the Mars Observer power subsystem during cruise, although a malfunction of this subsystem was given as a *potential* cause of the MO failure by the failure review board[15,16]. (See part III.)

Thermal Control

For TOPEX/POSEIDON, all spacecraft-component temperatures remained within their allowable flight limits during normal operating conditions except for the following. The temperature of the tape-recorder electronics exceeded the 50¡ limit and recorder operation procedure was changed to compensate for it. The high-gain antenna gimbal operating temperature showed an increase with time, but this was later limited by maximizing the travel of the Y gimbal in the negative direction.

There were no significant hardware thermal problems with Mars Observer.

Mechanisms

There were no problems with TOPEX/POSEIDON mechanisms.

Although the full deployment of the Mars Observer high-gain antenna was delayed, the delay had no significant impact on performance. There were no other mechanism problems.

Other Subsystems

All other TOPEX/POSEIDON and Mars Observer S/C subsystems experienced nominal performance during earth orbiting and Earth-Mars cruise, respectively. Possible catastrophic failures prior to the initiation of MOI in the power and pyro subsystems are discussed in Part III.

Payloads

The TOPEX/POSEIDON payload suffered single event upsets on all sensors resulting in some data losses. However, the upsets on all instruments have been within the design predictions and data loss has been less than expected.

Although the NASA altimeter (ALT) has been affected by SEUs, it has met all its performance requirements. The main concern, data loss from data not processed in real time, was alleviated by streamlining the procedure for faster ground recovery from SEUs. TWTA noise was reduced from 1dB to 0.2dB and is not a concern. Also not causing problems was the loss of five non-critical telemetry channels related to the switching of remote interface units discussed above.

The TOPEX microwave radiometer (TMR) has been operating nominally.

The TOPEX solid-state altimeter (SSALT) is operating nominally, though it had a late turn-on, and lost data due to satellite pointing errors and SEUs.

The TOPEX laser retroreflector assembly (LRA) has been operating with no problems.

The Doppler orbitography and radiopositioning integrated by satellite (DORIS) instrument is operating nominally, though it also had several initial minor problems due to procedural problems and SEUs.

The TOPEX global positioning system demonstration receiver (GPSDR) has collected 95-98% of the dual frequency data in spite of flight software errors. Two telemetry temperature-measurement channels were lost due to the remote-interface-unit failure, but this has had minimal impact on operations.

For Mars Observer, the only significant science, performed during cruise, involved the radio-science and resulted in successful completion of the gravity wave experiment. In addition, the KABLE experiment was successfully performed.

The Mars Observer camera had an astigmatism of the narrow-angle optics that caused image distortion which was mitigated by heating the camera mirror, and operating the camera at the point of best focus.

The only serious problem with the Mars Observer payload occurred with the magnetometer/electron reflectometer. This was the one instance, up until loss of S/C contact, where the MO S/C was not able to meet its requirements. The magnetometer suffered magnetic interference from the S/C caused by current loops resulting from the solar array operation. This compromised its ability to make dynamic (more serious of the two) and static magnetic field measurements of the Martian magnetic field. Because of the capability of making measurements of the Martian magnetic field during solar occultation, it was anticipated that most of the required data would have been obtained.

The case of the Mars Observer electron reflectometer (ER) was shorted to the S/C chassis ground. This would result in the loss of the lowest-energy-range electrons. The ER data was supplemental to the magnetometer data, providing additional data from the electron trajectories.

2.1.2 Summary of Performance, Anomalies and Work-arounds

Table 9 in Part III shows the number and percentage of the key functions that fall under each performance category for TOPEX/POSEIDON and Mars Observer. It also shows the number of relevant anomalies (total and per year of operation), and the types of fixes that were employed to mitigate the effects of anomalies.

Comparisons between TOPEX/POSEIDON and Mars Observer must be viewed in the light of the fact that, at the time of this investigation, the former had completed over 2/3 of its primary mission; while, Mars Observer had its mission terminated after 11 months, before its primary mapping phase. Prior to loss of contact, Mars Observer had percentages of key functions met without significant anomalies comparable to TOPEX/POSEIDON. Because of the MO loss-of-contact, the performance ratings in the tables are broken into two parts, one set for the 11 months of cruise and the other set for the point of failure forward.

If payloads are included (see Table 10 in Part III), it was known prior to the Mars Observer failure that the performance of the magnetometer/electron reflectometer was significantly affected by magnetic interference from the S/C. However, it was anticipated that useful data would still be collected during the primary mission.

2.1.3 Overall Spacecraft Performance

The overall S/C performance ratings for TOPEX/POSEIDON and Mars Observer are shown in Tables 11 (not inc. payloads) and 12 (inc. payloads) in Part III. Again, the performance for the two S/C is comparable up to the point of the Mars Observer failure. (The one exception is the problem with Mars Observer magnetometer which was to perform during its primary mission in Mars orbit). The rating for Mars Observer due to loss of contact prior to MOI is therefore a 5, total failure.

2.1.4 Mission Objectives Accomplished

Table 13 in Part III provides the mission objectives accomplished. TOPEX/POSEIDON has accomplished all its objectives to date. Mars Observer accomplished what it was expected to do during cruise, but never had a chance to accomplish any of its primary mission objectives beyond that.

2.1.5 Performance Met Through Redundancy/Margins/Work-arounds

The following lists the cases of performance met through various work-arounds and margin. Had these options not been available, performance would have been degraded or not met.

Redundancy:

TOPEX/POSEIDON: Remote interface unit, star tracker.

Mars Observer: Sun sensor, celestial sensor fan, NTO tank thermal control.

Resource/Design Margin:

TOPEX/POSEIDON: Remote interface unit expander channels lost due to ESD (rest of unit intact).

Mars Observer: Misaligned sun-sensor head, mitigated by use of multiple heads.

Operational/Software Work Arounds:

TOPEX/POSEIDON:

AACS needed flight software fix because gyro corrections not performed properly.

AACS caused NASA altimeter pointing error, requiring software patch.

AACS - sun sensor false sun presence readings required software fix.

Power subsystem - batteries charging anomaly, requiring change in procedure.

DORIS - bad satellite time correlation table, requiring using new synchronization sequence.

SSALT - mispointing fixed by software patch.

Thermal subsystem - HGA gimbal operating temperature was higher than expected, it was limited by maintaining the Y gimbal in a negative position for a long time.

Mars Observer:

Failed thermistor necessitated using alternate telemetry to monitor temperature of MO camera narrow angle electronics.

Distortion in narrow optics corrected by allocation of heat to mirror rim heater.

The PMIRR operated below required temperature, corrected by leaving auxiliary heater on.

Magnetometer interference from solar array should have been corrected by operation during solar occultation.

2.2 Performance Correlations

The correlation of subsystem and S/C performance with S/C and mission parameters requires a larger database of flight missions and S/C than available from the six missions covered in this and the first report of the FPC Task[1]. However, it is useful to consider the types of correlations that could be made with a larger database. The other JPL flight programs covered in the first report were Voyager 1 and 2, Galileo and Magellan, and one instrument, WFPC1. The reader is referred to that report for the data on those programs.

2.2.1 Correlation to Functional Performance

Table 14 in Part III provides a candidate set of parameters to be used in correlating performance with S/C and mission parameters. TOPEX/POSEIDON is the only S/C investigated so far that is not class A (WFPC1 was an instrument). However as a class B S/C its performance rating, a 2, was comparable or better than the class A S/C. The product assurance programs of both S/C covered here, JPL D-1489 equivalent with exceptions, was the same as Magellan, and weaker than the other S/C flight programs. Therefore, MO with the worst performance rating of all, a 5, provides a data point to correlate with the weaker product assurance programs.

Mars Observer was also one of three programs to be built in the system contract mode, a characteristic shared with Magellan. Related to this is the type of contract awarded, in this regard both TOPEX/POSEIDON and Magellan were cost plus contracts, while, Mars Observer was a fixed price contract with modifications.

In terms of the years of operation, TOPEX/POSEIDON had not completed its mission at the time of the study (2.3 years out of 3 for the primary mission), and the Mars Observer mission was terminated after only 11 months of the cruise phase. Therefore, neither one got as far into its mission (percentage of completion), as the four flight programs considered in the first report.

Both TOPEX/POSEIDON and Mars Observer were exposed to significant radiation environments, the latter to the earth's south Atlantic radiation anomaly and Mars Observer to solar proton events. However, the Voyagers were exposed to more intense radiation due to Jupiter's radiation belts, and Magellan and Galileo missions also took place during the upper end of the solar cycle.

As pointed out in the first FPC Task report[1], there are other parameters that were not tracked that may be relevant to the correlation of performance. One obvious characterization of a S/C is the state of the technology of the hardware for hardware ranging from electronic parts and boards to assemblies such as gyros. Since the JPL programs studied span a range of technology of about fifteen years, it was expected that significant changes in the state of technology have occurred in that time period.

In addition to the state of technology there is now a tendency to base new flight programs on inherited hardware design. TOPEX/POSEIDON was derived from a the GSFC Solar Maximum Mission Earth orbiting satellite, and the GSFC LANDSAT 4 and 5 Earth observational satellites, which were modular S/C built for shuttle retrieval. Mars Observer was derived from the GSFC TIROS/NOAA and Air Force DMSP meteorological satellites.

One parameter that covers a lot of territory is S/C complexity. There is no precise definition of S/C complexity, although it has been linked to things such as parts count, as well as, to an entire array of S/C capabilities and functions, and the complexity of the set of hardware performing each function. Table 3, in Part III, provides a listing of the types of S/C parameters that relate to complexity. The two S/C considered here have been rated as about equal in complexity to each other and to the Magellan S/C considered previously, but less complex than Voyager or Galileo.

In summary, TOPEX/POSEIDON and Mars Observer have more in common with the Magellan S/C than Voyager and Galileo. They represent later stages of technology, less complexity, and were performed in the system contract mode. In terms of ratings, Magellan and TOPEX were rated as well or better for the primary mission than any other S/C except Voyager 1. Magellan's extended mission was not highly rated, and TOPEX had not completed its primary mission as of the writing of this report. Mars Observer was rated the worst of all. Therefore, although these three S/C (MO, TOPEX/POSEIDON, and Magellan) constitute a trend in terms of characteristics, the three data points do not define a trend in terms of correlation of rating with relevant parameters.

2.3 Instruments

Tables 6a and 6b in Part III show the performance assessment of the instruments covered here. The TOPEX/POSEIDON instruments all received the highest rating, having accomplished their objectives fully up to the time of the study when the primary mission was 2/3 completed. Only the radio science on the Mars Orbiter had a chance to return actual science data; it successfully completed its part of a triad of radio experiments dealing with gravity waves. The only instrument on MO where problems were anticipated were the magnetometer/electron reflectometer. The magnetometer was impacted by magnetic background noise from the S/C solar array circuitry, but it was expected that data could be obtained during solar occultation and that the noise could be calibrated out. The electron reflectometer (ER), used to provide supplementary data to the magnetometer experiment on Martian magnetic fields, had lost part of its lowest-energy-range data capability due to shorting of the case to the S/C structure. Therefore, except for one part of the ER experiment, the actual performance of the MO instruments was not determinable.

2.4 Conclusions

TOPEX/POSEIDON and Mars Observer presented two situations significantly different from the flight programs reported in the first FPC report[1]. TOPEX/POSEIDON is an earth orbiting satellite, in contrast with the interplanetary S/C covered previously (except for the WFPC1 which was an instrument). However, TOPEX/POSEIDON shares a mode of operation with Magellan in being an orbiter instead of a fly-by. (The Galileo S/C is also an orbiter but that phase of the mission had not been initiated at the time of the writing of the report). The determination of success in meeting mission objectives is somewhat different for orbiters and fly-bys. Fly-by S/C have a limited opportunity for data capture and adjustment to unforeseen circumstances; while orbiters have an opportunity to gather enough data to make up for gaps in coverage. Complete loss of an instrument or sensor provides an equivalent situation for an orbiter and a fly-by. The Mars Observer situation was unique in that the S/C suffered a catastrophic failure before entering its primary mission phase. In addition, the cause of the catastrophic failure was not determinable and could only be described in terms of the most probable cause. Therefore, the only definitive assessment of S/C and subsystem performance involves the cruise phase. Aside from calibrations, the significant science performed during the cruise phase was the radio science gravity wave and KABLE experiments.

The TOPEX/POSEIDON S/C (with and without payload) received the highest possible rating for the 2.3 years out of 3 years of primary mission which was completed at the time of this study. It had also successfully fulfilled all of its mission objectives up to that point.

Except for not providing a magnetically clean environment for the magnetometer experiment, Mars Observer met all of its requirements up to the point of the pressurization of the bipropellant propulsion subsystem for Mars orbit insertion. There were the usual early-mission-phase difficulties with the ground support system. However, these were alleviated during the cruise phase and would not have impacted the ability of the S/C to carry out its primary mission. This is all that can be said with certainty about Mars Observer. The actual cause of the failure of the mission and the actual identity of the culprit subsystem or subsystems could not be deduced; only the most probable cause or causes of the mission failure within the time period of the loss of contact with the S/C could be investigated.

Part III: Individual Mission Reports

3.1 TOPEX/POSEIDON

3.1.1 Project Overview

3.1.1.1 Mission Description[2-5]

The TOPEX/POSEIDON (Ocean Topography Experiment) Mission is a joint effort between NASA and the French space agency (Centre National d'ƒtudes Spatiales Ð CNES). Its primary objective is to study the Earth's oceans by observing the global ocean circulation, as part of NASA's Mission to Planet Earth. The main goal of the TOPEX/POSEIDON mission is to understand the dynamics of ocean circulation and the role this circulation plays in climate change. It measures sea levels, maps basin-wide variations in currents, and monitors the effects of currents on global climate change. The measurements also allow scientists to study tides and waves, marine geophysics, and wind.

The TOPEX/POSEIDON spacecraft/satellite was launched into Earth's circular orbit on August 10, 1992, by a French 3-stage Ariane rocket from the European Space Agency's Space Center located in Kourou, French Guiana. The S/C mission and program characteristics are shown in Tables 1 and 2 respectively.

Following injection into the bias orbit, the Ariane 3rd stage oriented the satellite to its desired earth/sun pointing attitude and then enabled and fired the pyrotechnic devices to release the satellite. The separation springs in the adapter, which remained with the Ariane, imparted a separation velocity of 0.5 m/sec. The satellite travels in a 66¡ inclined orbit at an altitude of 1,336 km above the Earth's surface. This orbit allows coverage of 95% of the ice-free oceans every 10 days. The satellite's position is known to within 4 cm from Earth's center, an unprecedented accuracy that enables the satellite's radar altimeters to make precise measurements of changes in sea surface heights. It is scheduled to operate for a three year primary mission and three year extended mission.

The satellite carries six instruments. This mission is managed by JPL for NASA.

3.1.1.2 Spacecraft Description[2-7]

The TOPEX/POSEIDON S/C (Fig. 1), constructed by the Fairchild Space Company in Maryland, is a modification of the Multimission Modular Spacecraft (MMS) used for NASA's 1980 Solar Maximum Mission and the later Landsat-4 and -5 missions. The S/C systems characteristics are shown in Table 3. The 2400 Kg satellite carries six instruments housed in a special instrument module attached to the MMS satellite bus. The power is supplied by a single large solar-cell array. It is designed with a primary mission lifetime of 3 years, with an additional 3 years of extended mission operations.

In addition to the dish-shaped, High Gain Antenna (HGA) used for radar altimetry, the S/C has a variety of communication antennas to link the Mission with the NASA Tracking and Data Relay Satellite System (TDRSS), the DORIS tracking system, and the GPS navigational satellites.

Although classified as a Class-B flight, many of the critical functions have built-in redundancy. Redundancies are available for the instrument module interface unit to preserve the redundancy provisions of the mission sensors and instrument module support equipment. Redundancy is also employed for the electrical/Radio Frequency (RF) system to avoid catastrophic single-point failures.

The HGA and gimbal/boom assembly, including the deployment and pointing mechanisms, is mounted on the instrument module. The control and drive electronics for the HGA assembly are located in the instrument module. A single solar array is also mounted on the instrument module, together with a deployment mechanism, drive/slip ring assembly, and drive electronics.

The RF communication system is composed of redundant NASA standard TDRSS/GSTDN (Ground Spacecraft Tracking Data Network) transponder and RF components located in the Command and Data Handling (CDH) module and the instrument module. All elements of the data handling are fully redundant.

The pointing commands are from the On-Board Computer (OBC) in the CDH module. Redundant central processor modules are used for the satellite OBC, together with core memory units, giving a redundant storage capacity of 65536 18-bit words.

All elements of the Attitude Determination and Control Subsystem (ADCS) also incorporate redundancy, except for the Digital Fine Sun Sensor (DFSS) which is not the primary sensor for precision attitude determination. Redundancy has been incorporated into the Inertial Reference Unit (IRU) and star trackers, which are the primary attitude determination sensors.

The propulsion module is a hydrazine monopropellant propulsion system. It consists of four propellant tanks with bladder type propellant management devices. There are four 22-N thrusters for large delta-V maneuvers and twelve 1-N thrusters for small delta-V maneuvers and attitude control. The propulsion module has redundant thrusting capability for all orbit and attitude control functions. There are also redundant pressure transducers and six isolation latch valves. All thermostats and heater controls are redundant, and each redundant pair is capable of independent operation.

Telecommunications

The satellite RF telecommunication system operates with the Space-flight Tracking and Data Network (STDN) TDRSS and Deep Space Network (DSN) links. It is capable of periodic 2-way ranging (TDRSS only) and 1-way/2-way Doppler tracking (via TDRSS or DSN) for operational orbit determination. The HGA is used for normal TDRSS communications, and the Zenith and nadir antennas are used for special/contingency communications via TDRSS or DSN.

Normal satellite communications use the TDRSS multiple access or S-band single access communication services, including 1-way/2-way Doppler and ranging capabilities. In addition, the omni antenna is used for direct, 2-way contingency communications and 1-way/2-way Doppler with the DSN. Ranging is available via the TDRSS S-Band Single Access link.

The NASA 5-watt transponder receives and transmits telemetry data in the TDRSS and DSN modes, accommodates ranging data in the TDRSS mode, and 1-way/2-way Doppler in either mode.

The transmit and receive frequencies are 2287.5 MHz and 2106.4 MHz respectively. The Frequency Reference Unit (FRU) provides a stable timing reference at 19.056392 MHz. The system is capable of a 48 Kbps tape recorder playback rate and a high playback rate of 320, 384, or 512 Kbps, both at the Q channel, via the HGA and TDRSS subsystem assembly mode, or 768 or 1024 Kbps via the nadir omni and DSN. It also allows simultaneous real-time telemetry at 16 Kbps for all playback data rates of up to 512 Kbps.

Attitude Control

TOPEX/POSEIDON is a three-axis-stabilized satellite. Attitude control (nadir pointing) is maintained by reaction wheels, gyros, and magnet torquer bars for momentum unloading of the reaction wheels. Attitude determination is performed by using earth sensors, star trackers, magnetometers, and 3-axis gyroscopes.

The ADCS gives precise pointing knowledge and control of the NASA radar altimeter boresight along the nadir direction during normal on-orbit operations. The IRU and star trackers are the primary sensors for precision attitude determination, while the DFSS is a secondary sensor.

The ADCS also gives yaw angle (about nadir) knowledge and control to within one sigma error limits of 0.07 and 0.14 degrees, respectively, as required for solar array sun pointing, for pointing the HGA to TDRS, and for satellite modeling.

The satellite is capable of effecting large orbit change maneuvers and precision orbit trim maneuvers in any direction. All delta-V maneuvers are accomplished within a magnitude range from 1 mm/s to 15 m/s. Major orbit maneuvers are effected using the axial 22-Newton thrusters, while small drag makeup or trim maneuvers are accomplished using the axial 1-Newton thrusters. Attitude stabilization is obtained using attitude thrusters as required. The design of the satellite allows a total minimum delta-V of 160 m/s.

During in-plane orbit control maneuvers, the satellite orientation is kept aligned relative to a local vertical-velocity vector attitude (Z axis along the local vertical and axial thrusters aligned with the velocity vector). The ADCS also accommodates out-of-plane maneuvers.

Electrical Power

The power for TOPEX/POSEIDON is generated from the solar array and three batteries, which are capable of supporting an orbital average load power and associated battery charging power of 1018 Watts during normal mission operations. The Electrical Power System (EPS) supplies the mission sensors with a power bus voltage range of 23 to 35 Vdc, with typical satellite bus element loads at 28 Vdc. The batteries also provide a bus voltage of 28 Vdc.

The solar array is made up of four adjoining solar panels and, measures 8.7 m by 3.3 m. Three 50 Amp-hour Ni-Cd batteries, each with 22 cells in series, satisfy additional satellite power demands throughout the mission. The EPS provides unfused, redundant power buses to the standard internally-fused bus module and instrument module.

Command and Data Handling

Command and telemetry for all the satellite bus subsystems and mission sensors are controlled by the CDH Subsystem (CDHS) via the redundant satellite central unit, multiplex data bus, bus coupler units, and remote interface/expander units. The subsystem allows discrete commands and serial magnitude commands to execute in two modes: in the first mode, the command is executed when received (real-time mode); in the second mode, the command is executed at a predetermined S/C time (stored mode).

The CDHS also stores satellite telemetry data pending data transfer to the ground. It is capable of recording a complete, 16 Kbps telemetry stream at all times during the mission, including simultaneous data record and real-time telemetry, concurrent with stored data playback during TDRSS or DSN contact periods.

Three redundant tape recorders with storage capacities of at least 500 Mb each are available to ensure a continuous data acquisition and recording capability even with the loss of one tape recorder. Six PROM telemetry formats are available from the CDHS, including an OBC-controlled 16-Kbps flexible telemetry format that is reprogrammable by commands from ground control.

Propulsion

The propulsion subsystem consists of the propulsion module, which is attached to the satellite bus, and the auxiliary tank kit, which is located within the bus. It provides propulsion capability to support the attitude control subsystem via 3-axis reaction control during the initial acquisition and orbit maneuvers.

Major translation maneuvers for large orbit changes are carried out using the four axial 22-Newton thrusters, while small drag makeup or trim maneuvers are carried out using the twelve axial 1-Newton attitude/orbit control thrusters. These 1-N thrusters also provide effective attitude stabilization. All 16 thrusters use monopropellant hydrazine gas, and have redundant thrusting capability for all orbit and attitude control functions.

Thermal Control

The thermal control subsystem maintains operating and non-operating flight temperature limits during all mission phases for the satellite bus equipment and mission sensors. The flight temperature limits for the components are between 0 to 55 ¡C, with design temperature limits of -30 to +85 ¡C. Temperatures of the satellite bus and primary thermal surfaces are monitored via telemetry data from the ground. All thermostats and heater controls are redundant, and each redundant pair is capable of independent operation. The measured temperatures are used for performance verification and satellite modeling.

The emissivity and absorptivity of primary external surface materials are known to an accuracy of 0.03 for thermal blankets, louvers, radiators, and solar array. All the conducting surfaces of multilayer blankets are electrically grounded to the satellite structure to avoid electrostatic charging problems.

Computing and Software

Central Processor Modules (CPMs) and Core Memory Units (CMUs) are used for the TOPEX/POSEIDON satellite OBC. The OBC has a redundant storage capacity of 65536 18-bit words. It is capable of on-board fault detection and correction and/or status flagging for each critical satellite function.

Modified OBC Version-22 flight software developed for LANDSAT-5 was used as the baseline to meet the requirements for this mission. The OBC flight software is divided into two elements: the flight executive, together with its support applications processors, and the mission support applications processors. The flight executive controls software timing and execution, and calls upon application processors for specific software tasks.

Algorithms are included in the OBC software for satellite attitude determination and desired attitude acquisition and control. Control is also provided for requisite satellite yaw angle maneuvers to orient the solar array drive axis to the sun, and for executing orbit maneuvers as defined by ground commands.

3.1.1.3 Scientific Instrumentation[2-6]

The TOPEX/POSEIDON satellite carries six sensors: the NASA-supplied dual-frequency radar altimeter (this instrument is managed by Goddard Space Flight Center (GSFC) and was built by the Johns Hopkins University's Applied Physics Laboratory (APL)), Laser Retroreflector Array (LRA), TOPEX Microwave Radiometer (TMR), and experimental Global Positioning System Demonstration Receiver (GPSDR); and the CNES-supplied single-frequency, Solid-State Altimeter (SSALT), and DORIS dual-Doppler tracking system receiver. The instruments' program characteristics are shown in Table 4a.

NASA Altimeter (ALT)

The NASA dual-frequency altimeter is the primary instrument of this mission. The altimeter sends signals towards the ocean surface and uses the radar time-delay to determine significant waveheight. It operates at two frequencies: 13.6 and 5.3 GHz (corresponding to Ku- and C-band respectively). Dual-frequency operation permits correction for the ionospheric-electron delay effects. This instrument is fully redundant and incorporates well understood, flight-tested technology.

<u>TOPEX Microwave Radiometer (TMR)</u>

The companion microwave radiometer, developed by JPL, operates at frequencies of 18, 21, and 37 GHz to estimate the total atmospheric water-vapor content. The 21 GHz channel is the primary measurement channel; 18 and 37 GHz channels are used to remove the effects of wind speed and cloud cover, respectively. These data allow reduction of the water-vapor delay error to 1 cm, thereby permitting an overall altimetric precision of 3 cm.

<u>Solid-State Altimeter (SSALT)</u>

The advanced, experimental solid-state Poseidon altimeter, designed by CNES and built by Alcatel Espace, shares the same antenna as the NASA altimeter, but operates at a single frequency of 13.6 GHz. The ionospheric-electron correction is obtained from a model that makes use of the simultaneous dual-frequency measurements of the DORIS tracking system. It provides similar performance to the NASA altimeter, though the telemetry data rate is a factor of 7 less because of more extensive on-board processing.

<u>Laser Retroreflector Assembly (LRA)</u>

The NASA LRA tracking system (managed by GSFC) operates by laser ranging to an array of reflectors (built by APL) around the altimeter antenna, which permit the satellite to be intermittently tracked by a worldwide ground-based network of 12 laser stations within about 2 cm of known position. These data will be used with computer models of the Earth's global gravity field for precision orbit determination and calibration of the altimeters.

<u>Doppler Orbitography and Radiopositioning Integrated by Satellite (DORIS)</u>

The DORIS Doppler receiver system determines the satellite's velocity by measuring the Doppler shifts of two ultrastable microwave frequencies (2.036 GHz and 401 MHz) transmitted by a global network of some 50 ground-based beacons whose positions are known within a few centimeters. This receiver system is managed by CNES and manufactured by Dassault Electronique (receiver), CEIS Espace (beacons), and CEPE and OSA (quartz oscillators).

Global Positioning System Demonstration Receiver (GPSDR)

The experimental GPS receiver, developed by Motorola under contract to JPL, is used to demonstrate GPS capabilities. This system allows continuous S/C tracking with an estimated accuracy of 10 cm or better.

3.1.1.4 Project Management

The TOPEX/POSEIDON Mission is managed by the Jet Propulsion Laboratory for NASA's Office of Mission to Planet Earth. The Poseidon part of the mission is managed by the French Space Agency, CNES. Fairchild Space Co. is the prime contractor for the S/C bus. CNES was responsible for the development of 2 instruments (DORIS and SSALT), the Ariane launch vehicle, and the control and data processing of the French instruments. C. A. Yamarone Jr. is the project manager and Dr. Lee-Leung Fu is the project scientist.

3.1.2 Performance[8-11]

3.1.2.1 Spacecraft Performance

TOPEX/POSEIDON was launched into orbit on schedule, with no countdown problems. The ascent trajectory, orbit injection, and separation from the Ariane launch vehicle were all normal. So far, with more than two-thirds of the primary mission completed, the satellite has encountered few hardware anomalies. Most of them (56%) have been due to ground software or human errors. The majority of these errors had minimal risk to the mission. Table 5a tabulates, in matrix form, the performance assessment of the spacecraft's key functions.

To date (November, 94) there have been three hardware failures:

(1). Advanced star tracker ASTRA1-B.

(2). Remote interface unit RIU-6A passive analog telemetry.

(3). Remote interface unit RIU-6B expander unit passive analog telemetry.

The following lists some of the significant anomalies encountered during the first few critical months of operation of the satellite. However, most of them were minor and were corrected by software patch or procedural changes (the more significant anomalies that were serious enough to turn into a problem/failure report (PFRs) are detailed in later sections):

(1). Switching to Mode 4B in ADCS was deferred due to a 3¡ roll axis error (8/11/92). ADCS was kept in Mode 4A using the Mode 2 controller. ADCS was successfully sequenced into Mode 4B after a software patch on OBC to correct roll error (8/14/92).

(2). Solar array switch to OBC control was only successful on second attempt because of a procedural error on first attempt (8/15/92).

(3). Orbit calibration maneuver did not occur due to a Project Operations Control Center (POCC) timing error (8/17/92). The maneuver was successfully accomplished after correcting for a timing error (8/20/92) and the thruster burn was within 4% of predictions.

(4). Initial GPSDR turn-on and memory load operations were completed (8/19/92), but tracking was not successful due to a software error. The modified software was loaded (8/24/92). Initial operations were satisfactory but subsequent problems were also encountered due to software errors and constellation outages.

(5). Initial ALT operations were unsatisfactory due possibly to a Single Event Upset (SEU) in the ALT computer (8/25/92). Operation was satisfactory after ALT reinitialization (8/26/92).

(6). The orbit inclination maneuver was successful, but the satellite entered the safehold mode about 3 sec before burn termination, as a result of improper setting of roll fault detection and correction limit (8/27/92).

(7). ALT was hung up prior to the inclination maneuver (8/27/92). After the maneuver, ALT was autonomously switched to idle when the satellite entered the safehold mode. Satellite recovered from the safehold mode on 8/28/92.

(8). ALT experienced the first of numerous SEUs (9/3/92). It self-recovered from most of the subsequent SEUs.

(9). GPSDR hung up for the first of several times (9/9/92) due to GPSDR flight software problems.

(10). ADCS entered safehold mode due to a yaw rate gyro scale factor error (9/23/92). The factor error was corrected with an uplinked table load (9/24/92).

(11). ALT and SSALT had a 0.3¡ misalignment in pitch and 0.08¡ in roll (10/1/92). Correction was made (10/13/92) by the uplinked modified antenna pointing bias values (pitch = +0.3¡; roll = -0.08¡).

(12). There were spikes in the ADCS ephemeris processing data (10/7/92). They were corrected by software work-arounds on the ground.

(13). Advanced star tracker ASTRA1-B failed to acquire stars (11/25/92), apparently due to a SEU.

Table 5a shows the performance assessment of the spacecraft's key functions. The S/C had a number of anomalies, many of which were overcome by software patches and other work-arounds. The following paragraphs detail the subsystem performance and pertinent anomalies.

Telecommunications

So far, telecommunications performance requirements have been met with only occasional disruptions in the communications link. NASA communications network (NASCOM) data line problems were essentially the source of all non-recoverable 16 kbps real time telemetry data losses.

There was some problem with transmitter B. Occasional RF interference on the XMTR-A downlink occurred due to transmitter XMTR-B being turned on and caused a momentary loss of lock. XMTR-B was initially logged on for additional contingency following launch. It was later powered off. Transmitter XMTR-A is now kept continuously on and XMTR-B used only for emergency or DSN passes.

The overall performance of the FRU in time correlation is being met, with radiation the primary perturbator of frequency. Time correlation using TDRSS is currently estimated to have an uncertainty of ^{2}10 microseconds.

Command and Data

The command and data subsystem has experienced two hardware failures in the remote interface units, and several initial minor problems.

The OBC encountered some difficulties in its initial dump after ground system reboot (8/18/92). In that incident (PFR#59027), status buffer messages were not being stored and received through downlink telemetry. An uplinked software patch correctly reset 2 parameters in the interrupt (8/18/92). Both hardware and software in the OBC are currently performing well.

In another incident (PFR#59029) a telemetry monitor (TMON) algorithm turned off the NASA altimeter for load shedding, when the satellite entered its first eclipse season (8/30/92). The problem was corrected with an uplinked software patch (1/21/93). However, the TMON algorithm must still be disabled for commanding (e.g. for orbit maintenance maneuvers) when there is a significant drop in current due to battery discharge from power sharing. Operational procedures are in place to handle these instances.

Servo error variations, changes in AC excursions, and shift in DC characteristics were observed with all 3 tape recorders during record cycles. These events did not occur during playback cycles, and appear to have no bearing on tape speed regulation or playback data rate. Over 99.9% of all taped data have been received and processed. A new intermittent spot on tape recorder TR-C had numerous random data gap occurrences (0.15% at worst), which were not believed to be a problem of the tape media, NASCOM, RF link, ground control or TDRS. SEUs have occurred on the CU-B standby command counter, but they had negligible impact on the mission.

Remote Interface Unit hardware failures

Both sides of the remote interface unit RIU-6A and RIU-6B have suffered hardware failures. RIU-6A reported erroneous Passive Analog (PA) telemetry readings (PFR#59033) on June 4, 1994. Twelve of the thirteen failed channels are for thermal measurements, with the remaining channel reporting HGA boom position. All readings except for one reported erroneously high values (>200), while the other channel remained pinned at zero. This was likely due to electrostatic discharge induced failure of the multiplexer in RIU-6A. The remote interface unit was switched to the B-side RIU-6B on July 6, 1994. It was found that RIU-6B had a similar electrostatic discharge induced failure, but only to the Expander Unit (EU) telemetry input channels (PFR#59033). Most expander unit channels indicted a maximum or near maximum values, while one remained pinned at zero. The RIU-6B command I/F and all other telemetry points were functioning properly. As the important HGA gimbal temperature measurements were valid through RIU-6B, it was elected to remain configured on RIU-6B indefinitely. These failures did not affect real time telemetry significantly.

Attitude and Articulation

The Attitude Determination and Control Subsystem (ADCS) had one hardware failure and several instances where satellite orientation was temporarily lost. These instances of problem and failure are outlined below.

The roll axis propagated from about 0.2¡ to 4¡ (PFR#59026) shortly after launch (8/14/92). Real-time analysis indicated that the flight software was not performing gyro corrections properly and the large attitude errors were caused by incorrect advanced star tracker scale factors in the flight software. A software patch was uplinked on 8/14/92 to correct the scale factors.

Large spikes in the Earth Sensor Assembly (ESA) position data were observed in both ESAs (8/19/92). They were large enough to saturate the fine telemetry channel (>5¡) (PFR#59028). These spikes were caused by SEUs and strongly correlated with the South Atlantic Anomaly environmental effect. They do not appear to represent a significant danger to the ESAs. Fault detection and control limits were widened slightly, and the spikes have had no impact on the mission.

The NASA altimeter was initially seeing a pointing error of approximately 0.6¡ (PFR#59030, 10/23/92). Two flight software patches were uplinked on 10/1/92 to reduce pointing error to 0.1¡-0.2¡. Further refinements were made to satellite pointing via matrix updates and altimeter boresight calibrations.

The ASTRA1-B advanced star tracker experienced a failure on 11/25/92 (PFR#59031). Fault detection was initiated when the tracker failed to acquire stars after three consecutive orbits. The tracker is currently off-line, with the shutter closed, due to a perceived high background count (bright body indication). Action is still pending (PFR is not closed). The current attitude knowledge is being adequately maintained by ASTRA1-A and the digital fine sensor.

The Digital Fine Sun Sensor (DFSS) experienced a false sun presence reading (12/30/92) from glint reflections within the DFSS field of view (PFR #59032). An uplinked software patch solved that problem. It allowed the sensor to test for and validate actual sun presence. Only valid sun presence readings are now processed by the DFSS. False sun presence readings have not reoccurred (as of 5/11/93). The DFSS also outputs erratic data in the field of view region between -17¡ to -22¡ (it's field of view is ±32¡). This could be a reflection of some kind, and it does not cause measurable attitude degradation.

The ASTRA1-B star tracker Thermoelectric Cooler (TEC) was powered off for no apparent reason during a lunar interference that opened its shutter (11/7/93). The TEC was commanded back on during the next lunar interference (12/24/93) and has operated normally since then.

The gyro scale factor (pitch and yaw) is increasing by 0.00007% per day which is still within the stability specification of ±0.005% in 30 days with ±0.5% maximum. The pitch and yaw gyro scale factors show a slight but distinct trend. A new ADCS calibration parameter upload has been made to compensate for the scale factor changes.

Gyro jitter caused by solar array resonance has been observed, which is correlated with high b' periods. This jitter does not adversely affect pointing or the solar array.

Electric Power

The electric power subsystem has met all its power requirements. Peak charge currents have been maintained below 20A, with the average not allowed to decrease below 13 A (accomplished by offsetting the solar array). Battery temperatures have been controlled and maintained at 5 ¡C. The charge/discharge ratios and end-of-night voltage degradation have been maintained within acceptable limits, and the batteries are performing nominally.

The solar array and drive electronics are performing consistent with pre-launch predictions. The standard power regulator unit output has shown no degradation since launch. Its efficiency has consistently been around 92-93 %.

However, solar array off-pointing has been adjusted several times to maintain constant current. It has been changed, since launch, from a $+57.5^o$ to $+53^o$ offset to raise the average battery charge current to counterbalance normal solar array degradation.

Propulsion

No problems have been encountered with the propulsion subsystem. Orbit maintenance maneuvers have been nominal. The initial rate damp and wheel unloads were highly satisfactory and not required thereafter. However, several of the propulsion subsystem components and modes of operation have remained untested in orbit.

Two propellant tank pressure transducers, which should have similar values, have been diverging with time. The two transducers were reported to differ by 6 psia (7/26/94). The drift is linear with time and one transducer is reading high and the other low. The average value of the two has been used for maneuver calculations.

Thermal Control

Thermal control has been satisfactory for all modes and flight conditions. During normal operating conditions, all S/C components have remained within their allowable flight limits (except for the tape recorder and HGA gimbal, but none were serious).

During record and playback cycles, the temperature of the tape recorder A and C electronic unit reached 53.44 ¡C, exceeding the required limit of 50 ¡C. The high limit was then raised to 55 ¡C, and recorder operation procedure was changed to compensate for it.

The HGA gimbal operating temperature has apparently increased over time. There was a correlation between the gimbal temperature and the Y gimbal position. A change in gimbal protocol was demonstrated to have a positive or negative effect on the temperature. The operating temperature of the gimbal is now limited by maximizing travel of the Y gimbal in the negative position.

Other Subsystems

All other subsystems have experienced mostly nominal performance as shown by Table 5a.

3.1.2.2 Payload Performance

Single event upsets have been observed on all sensors (ALT, SSALT, DORIS, and GPSDR), and all have experienced data losses resulting from them. Upsets for all instruments have been within the design predictions and data loss has been less than expected. The instruments' performance assessment is summarized in Table 6a. The following is the performance of each of the instruments.

NASA Altimeter (ALT)

All performance requirements are being met by the instrument. Side B of the instrument has not yet been turned on. The NASA altimeter has been operating nominally despite numerous SEUs which were anticipated. There have been over a dozen self-correcting SEUs (with approximately 3.5 days of data lost), 3 safeholds, and 3 ground reinitializations since launch. A new streamlined procedure for faster ground recovery from SEUs has been developed to minimize data loss. Science data loss from all causes in the first year was ~2%, and it was reduced to ~1.1 % in the second year. The data returned is exceeding the overall mission requirement of 81% of over-ocean data. The noise level for one-second measurement in both Ku- and C-bands are all within specification for waveheights 2m, 4m, and 8m.

The Traveling Wave Tube Amplifier (TWTA) had a transmission power noise of ~1dB earlier in the mission. It has now been reduced to ~0.2 dB or less. This noise has no effect on the quality of the data.

ALT lost five non-critical telemetry channels, mainly in temperature measurements, when the failed remote interface unit RIU-6A was switched to the backup RIU-6B unit.

TOPEX Microwave Radiometer (TMR)

The TMR has been operating nominally. No commands have been required since its initial turn-on. It has been in continuous operation within its calibrated temperature range. All engineering parameters are normal.

Solid-State Altimeter (SSALT)

The first turn-on of the instrument was unsuccessful. The problem was solved by performing a new turn-on sequence. After launch, satellite pointing errors had a significant effect on the data. SSALT has suffered four SEUs with several days of data lost. All SEUs were over the South Atlantic Anomaly Region.

SSALT is now operating nominally. Currents, voltages and temperatures are all nominal. Altitude noise is within specification. The mean data noise has been reduced from ~2.4 cm to ~1.8 cm with software changes. No problems have been encountered for data acquisition over the sea. The instrument has been operational for 69.9 days (as of July 1994), with 62.2 days of valid science data (a ratio of 89.6%).

Laser Retroreflector Assembly (LRA)

No problems are reported for the Laser Retroreflector Assembly. Although detailed performance information is not readily available, the network of laser stations are all returning excellent ranging data which are used for precise orbit determination.

Doppler Orbitography and Radiopositioning Integrated by Satellite (DORIS)

The first DORIS synchronization had a bad satellite time correlation table during command generation, and was corrected by using a new synchronization sequence. DORIS has suffered 7 SEUs (as of 7/21/94) over the South Atlantic Anomaly Region, many requiring reinitialization. Several beacon data loads from JPL were also required, but they had no significant consequence on orbit precision.

DORIS is now operating nominally. The currents, voltages and temperatures are all nominal. The mean current noise of the Doppler measurements is ~0.5 mm/s, and typical coverage was 77% of the time in the first year and 80-85% in the second year, with the 50 DORIS beacons worldwide. The orbit precision in the CNES operational orbit is 20-30 cm rms radial and <3 cm rms radial in the CNES precise orbit.

Global Positioning System Demonstration Receiver (GPSDR)

GPSDR has experienced a number of flight software errors; however, it has collected 95-98% of the dual frequency data. Two telemetry channels (temperature measurements) were lost when the failed remote interface unit RIU-6A was switched to the backup RIU-6B unit. The flight receiver performance is meeting expectations with the on-board clock offset usually good to 1 ms or better. The root mean square altitude is good to 3 cm or better using dual frequency data in comparison with the GPS precision orbit determination. The tracking coverage has exceeded requirements, with only 14 days lost (non-tracking days) due to errors and problems. Performance is within specification.

3.1.2.3 Mission Objectives

The primary science objectives of this mission have been met so far (see Table 7a for a list of the objectives of the mission). With two-thirds of the mission completed the satellite is performing well, with few problems and anomalies. From launch through 7/15/94, there have been 9 PFRs and 3 flight hardware failures reported. There were also 419 ISAs (Incidents/Surprises/Anomalies Reports) written, 392 of which have been resolved and closed. 56% of these ISAs were due to ground software or human errors that were corrected by software patches or procedural updates. There were numerous SEUs occurring in many subsystems with small amounts of data loss. The majority of the

ISAs were of minimal risk to the mission. Many of the redundancies have still not yet been used. All systems are working within expectations.

3.1.3 Summary of Results

Table 8 summarizes the S/C performance by key functions. Table 9 and 10 give the number of anomalies and the various work-arounds that have been used to fulfill the mission requirements, with and without the payloads respectively. The overall performance rating for this S/C is shown in Tables 11 and 12, with the payloads included and excluded respectively. The degree to which the mission objectives are accomplished is shown in Table 13. Table 14 shows examples of several possible correlations to functional performance and anomalies. Class and Product Assurance Program are obvious correlations that would affect the overall performance since they have an impact on the amount of risk in the design (whether redundancy is available) and its verification. Another interesting correlation is the type of implementation mode used, whether it is developed in-house or subcontracted.

3.1.4 Conclusions

TOPEX/POSEIDON is well on its way of achieving its primary mission objectives of 3 years. Thus far, there have been few problems and anomalies. The 3 hardware failures occurred in the satellite were overcome by redundancy, operational work-arounds or design margin. The project has met all its science objectives. This mission has been very successful, giving valuable ocean topography data and current circulation information to the scientific community.

3.2 MARS OBSERVER

3.2.1 Project Overview

3.2.1.1 Mission Description[12-14]

Mars Observer was a NASA global mapping mission to study the surface, atmosphere, interior, and magnetic field of Mars. The scientific mission was to last for one Martian year (equivalent to almost 669 Mars days, or 687 Earth days), and this would have allowed the S/C to examine the planet through its four seasons. Its science objectives were to:

1) Determine the global elemental and mineralogical character of the surface material;
2) Define globally the topography and gravitational field;
3) Establish the nature of the magnetic field;
4) Determine the time and space distribution, abundance, sources, and sinks of volatile material and dust over a seasonal cycle;
5) Explore the structure and aspects of the circulation of the atmosphere.

The global studies of the planet's geology and atmosphere were intended to give scientists information about the planet's evolution. This mission was to provide scientists with a global portrait of Mars that would help planetary scientists to better understand the history of Mars' geology and climate, and provide clues about the planet's interior and surface evolution.

Mars Observer's scientific instruments were intended to examine Mars in detail from above the atmosphere. Collectively, the instruments should have covered much of the EM spectrum. Each instrument should have produced sets of data that would contribute to a wide variety of scientific investigations.

The Mars Observer S/C was launched on September 25, 1992, aboard a Titan III rocket at Cape Canaveral, Florida. Its mission and program characteristics are shown in Tables 1 and 2 respectively. The expendable commercial launch vehicle (Martin Marietta Commercial Titan, Inc., Denver, Colo.) carried it into Earth orbit. From there, the Transfer Orbit Stage (developed by Orbital Sciences Corporation of Vienna, Va.) boosted the S/C into an interplanetary orbit leading to Mars.

It took an 11-month cruise to arrive at the Red Planet. The S/C was planned to have been placed, initially, in a large elliptical orbit around the planet. The orbit should have gradually been adjusted to a near-circular, 400 km orbit, inclined 93¡ to the planet's equator, where the S/C would fly regularly over the Martian poles about every two hours. This process should have taken about four months. This mapping orbit was planned to be sun-synchronized at 2 p.m. (sunlight at the same angle on the day side throughout the mission).

In addition to the spacecraft-based scientific program, Mars Observer was scheduled to participate in an international Mars investigation, the Russian Mars '94 and '96 Missions, through an agreement with France and the Russian Commonwealth of Independent States (CIS). The Russian plan was to launch two S/C, one in 1994 and the other in 1996. The first was to deploy penetrators into the surface of Mars and land small instrument packages on the surface for direct sampling of both the atmosphere and the surface. The Mars Balloon Relay Experiment was scheduled to involve using its relay equipment, consisting of a transmitter-receiver, to periodically receive and relay scientific and engineering data from these landed packages. In 1996, the Russian plan was to launch instrument packages, a balloon and, perhaps, a surface rover that would relay information back to Mars Observer in late 1997.

3.2.1.2 Spacecraft Description[12-14]

The Mars Observer S/C (Fig. 1) used, where possible, existing designs developed for Earth-orbiting satellite missions to minimize the cost of design, development and fabrication. The S/C was based on the electronic architecture of General Electric weather satellites TIROS/NOAA and DMSP (Defense Mapping Satellites Program), modified for the Mars mission. Most electronic subsystems used proven designs from previous satellite applications. Mars Observer was built under contract with NASA and JPL by General Electric Astro-Space Division (now part of Martin-Marietta Corporation) in Princeton, New Jersey. The S/C had a three-year design lifetime and was equipped with one large solar array, consisting of six solar panels. With its fuel, Mars Observer and its science instruments weighed approximately 2573 Kg. (The S/C systems characteristics are shown in Table 3.)

At launch, the spacecraft's main communication antenna, instrument booms, and solar array were folded close to the S/C bus, which was box-shaped. Very early in the cruise phase, these structures were partially extended. Four of the six solar panels were exposed. The main communication antenna was raised on a 6-m boom. (The solar array should have fully unfolded after the S/C reached its mapping orbit around Mars and deployed away from the bus). The two 6-m instrument booms carried two of Mars Observer's seven instruments: The Magnetometer/Electron Reflectometer and the Gamma Ray Spectrometer, which were partially extended.

Telecommunications

The telecommunications design of Mars Observer was capable of simultaneous radiometric tracking, telemetry, commanding, and data acquisition through the Deep Space Network (DSN). The 34-m high-efficiency subnetwork of the DSN provided daily uplink and downlink communication with the S/C at X-band frequency, while the 70-m antenna network also provided periodic Very Long Baseline Interferometry (VLBI) and real-time, high-rate telemetry and radio science support to the mission.

The S/C could also accommodate a Ka-band link experiment (KABLE) that generated a coherent downlink carrier at 4 times the X-band downlink frequency. The non-mission-critical KABLE spacebourne-hardware had an output of 50 dBm.

The main communication antenna was the 1.45-m diameter parabolic High Gain Antenna (HGA), supported on a 6-m boom for a clear view of Earth. Engineering telemetry from the bus contained information that should have been able to reconstruct the HGA pointing to an accuracy of 3 mrad (3s). There were also three low-gain antennas for communications (2 for receiving and 1 for transmitting).

Mission operation was conducted at JPL using the Advanced Multimission Operations System. During its 687-day mapping cycle, Mars Observer was scheduled to return more than 600 Gbits of scientific data, more than that returned by all previous missions to Mars.

Attitude Control

The S/C was capable of automatically maintaining its orientation and stability during all phases of the mission using horizon sensors, a celestial sensor, sun sensors, gyroscopes, and reaction wheels. The horizon sensor, adapted from a terrestrial design, was intended to continuously locate the horizon and provide signals to the S/C during the orbital mapping period. The celestial sensor assembly was used for attitude reference during the 11-month cruise and planned to be used as backup during mapping orbits.

The S/C was able to control the pointing of the base plate (nadir panel) of the body-mounted science instruments to within ±10 mrad (per axis, 3-s) with respect to the orbital reference coordinate system. The S/C also provided sufficient engineering telemetry in the data streams to obtain knowledge of the base plate of the body-mounted instruments' pointing (after-the-fact, non-real-time-reconstruction) to within ±3 mrad (per axis, 3-s).

In its circular mapping orbit, the S/C was planned to rotate once per orbit to keep the instruments pointed at the planet.

Electrical Power

The S/C was capable of supplying, controlling, converting, and distributing all electrical power required for S/C and payload functions. The power subsystem could support normal S/C operations in a Mars orbit with the worst-case eclipse.

The S/C power came from the solar array and two 42-AHr Ni-Cd batteries. The batteries were charged by the spacecraft's large solar array (3.7 m by 6.5 m), consisting of six solar panels, which could generate more than a kilowatt of power at a voltage of 28 Vdc ±2%. Once during each 118-minute orbit, the S/C was expected to enter Mars' shadow for about 40 minutes and would rely solely on battery power.

Command and Data Handling

Spacecraft and instrument control was accomplished by onboard microprocessors and solid-state memories. Scientific and engineering data were stored on tape recorders (total capacity 138 Gbit) for daily playback to Earth. Additional tracking time should have allowed information to be returned in real time from selected instruments whenever Earth was in view.

The spacecraft's Command and Data Handling Subsystem (CDHS) had redundancy and cross strapping features. The S/C could process two classes of commands: real time commands (RTC) and stored sequence commands (SSC). RTC allowed an immediate action by the S/C bus as determined by the operation code within the command, while SSC were stored (storing capacity 1500 16-bits) in the on-board sequence memory along with associated time tags. Combined science and engineering data stream could be returned in real time or recorded for later playback.

The S/C timing reference provided an unambiguous, binary count to the Payload Data Subsystems (PDS) every 1 sec and a timing pulse every 125 ms. The time supplied was capable of being correlated with a known epoch to within 20 ms. The stability of the clock frequency source was such that the total drift over a 21 day period was predictable to an accuracy of 20 ms.

Propulsion

There was a total of 24 thrusters aboard the S/C. Four 490-N and four 22-N bipropellant thrusters were designed for major translation maneuvers and Mars orbit insertion, eight 4.5-N thrusters for trim maneuvers, and eight 0.9-N thrusters for momentum unloading and steering. These smaller thrusters also provided effective attitude stabilization. The monopropellant systems used hydrazine gas, and the bipropellant systems used monomethyl hydrazine (MMH) and nitrogen tetroxide (NTO).

The estimated delta-V requirement from injection through mapping orbit insertion was 2249 m/s; and 45 m/s for mapping orbit maintenance. An additional 12 m/s was budgeted to raise the S/C to the quarantine orbit at the end of the mapping phase. The total propellant weight carried by the S/C was 1346 kg.

Thermal Control

The payload was thermally integrated with the S/C thermal design to maintain the temperature of each instrument housing and outer radiator shields within the instrument temperature limits of -20 to +30 ¡C operating, and -30 to +40 ¡C nonoperating, with the nominal temperature being 25 ¡C.

Blankets, louvers, and heaters were used for thermal control. The temperature of each instrument was monitored with sensors attached to the mounting surface. The temperature could be read out through the bus telemetry.

The gamma ray spectrometer (GRS) and pressure modulator infrared radiometer (PMIRR), which required cooling of their detectors, had their own radiative coolers. The PMIRR radiator was located adjacent to the PMIRR instrument, and the GRS radiator enveloped the GRS sensor head.

3.2.1.3 Scientific Instrumentation[12-14]

Mars Observer carried seven instruments to gather scientific data (the instruments' program characteristics are shown in Table 4b):

1) A gamma ray spectrometer, to measure the abundance of elements on the surface of Mars.
2) A thermal-emission spectrometer, to map the mineral content of surface rocks, frosts and the composition of clouds.
3) A line-scan camera, to make low-resolution images of Mars (for studying the climate), and medium- and high-resolution images of selected areas (for studying surface geology and interactions between the surface and the atmosphere).
4) A laser altimeter, to determine the topographic relief of the Martian surface.
5) A pressure-modulator infrared radiometer, to measure dust and clouds in the atmosphere; and to produce profiles of temperate, water vapor, and dust opacity.
6) Radio-science equipment, to measure atmospheric refractivity to determine the temperature profile of the atmosphere; and to measure the gravity field of Mars using the tracking data .
7) A magnetometer and electron reflectometer, to determine the nature of the magnetic field of Mars and its interactions with the solar wind.

Gamma Ray Spectrometer (GRS)

The Gamma Ray Spectrometer was used to characterize the chemical elements (e.g., uranium, thorium, potassium, iron, and silicon) present on and near the surface of Mars within an area of a few hundred kilometers in diameter. The data were to be obtained by measuring the intensities of gamma rays emerging from the Martian surface. These high-energy rays are created from the natural decay of radioactive elements or caused by cosmic rays interacting with the atmosphere or surface. By observing the number and energy of these gamma rays, it is possible to determine the chemical composition of the surface. The instrument development was managed by NASA's Goddard Space Flight Center.

Thermal Emission Spectrometer (TES)

The Thermal Emission Spectrometer was useed to measure infrared thermal radiation emitted from the Martian atmosphere and surface. From these measurements the thermal properties of Martian surface materials and their mineral content should have been determined.

The spectrometer (a Michaelson interferometer) was useed to determine the composition of the surface rocks and ice, and map their distribution on the Martian surface. The instrument was intended to investigate the advance and retreat of the polar ice caps, as well as the amount of radiation absorbed, reflected and emitted by these caps. The distribution of atmospheric dust and clouds was planned to be examined over the four seasons of the Martian year by the spectrometer.

Mars Observer Camera (MOC)

The camera system was used to photograph the Martian surface with the highest resolution ever attained by an orbiting civilian S/C. Low-resolution global images of Mars was scheduled to be acquired each day using two wide-angle cameras operated with 7.5 km resolution per pixel. These cameras should have acquired moderate-resolution images at 1.4 m per pixel for features of special interest.

The low-resolution camera system was intended to capture global views of the Martian atmosphere and surface to allow scientists to study the Martian weather and related surface changes on a daily basis. Moderate-resolution images were planned to be used for monitoring changes in the surface and atmosphere over time. The high-resolution camera system was planned to have been used very selectively because of the high data volume required for each image.

Mars Observer Laser Altimeter (MOLA)

The Laser Altimeter used a very short pulse of light emitted by a laser to measure the distance from the S/C to the surface with a precision of several meters. These measurements of the topography of Mars were intended to provide a better understanding of the relationship among the Martian gravity field, the surface topography, and the forces responsible for shaping the large-scale features of the planet's crust.

Pressure Modulator Infrared Radiometer (PMIRR)

The radiometer was used to measure the vertical profile of the tenuous Martian atmosphere by detecting infrared radiation from the atmosphere itself. For the most part, the instrument was intended to measure IR radiation from the limb, or above the horizon, to provide high-resolution (5 km) vertical profiles through the atmosphere.

The measurements were intended to be used to derive atmospheric pressure and determine temperature, water vapor, and dust profiles from near the surface to as high as 80 km above the surface. Using these measurements, dynamic models of the Martian atmosphere, including seasonal changes that affect the polar caps, should have been constructed and verified.

Radio Science (RS)

The radio science investigation would have used the spacecraft's telecommunication system and the giant parabolic antennas of NASA's DSN to probe the Martian gravity field and atmosphere. These measurements were intended to help scientists determine the structure, pressure, and temperature of the Martian atmosphere.

During the part of the orbit when the S/C was in view of Earth, precise measurements of the frequency of the signal received at the ground tracking stations were scheduled to be made to determine the velocity change (using the Doppler effect) of the S/C in its orbit around Mars. These Doppler measurements, along with measurements of the distance from Earth to the S/C, were intended to be used to navigate the S/C and study the planet's gravitational field. Gravitation field models for Mars were planned to be used along with topographic measurements to study the Martian crust and upper mantle.

Magnetometer/Electron Reflectometer (MAG/ER)

In addition to searching for a Martian planetary magnetic field, the Magnetometer/Electron Reflectometer would have been used to scan the surface material for remnants of a magnetic field that might have existed in the distant past. The magnetic field generated by the interaction of the solar wind with the upper atmosphere should also have been studied.

3.2.1.4 Project Management

The Mars Observer mission was managed by the JPL for the Solar System Exploration Division of NASA's Office of Space Science. Astro-Space Division of General Electric in Princeton, New Jersey (now part of Martin-Marietta Corporation), was the prime contractor for the S/C. Mars Observer's project managers have included William I. Purdy Jr., David D. Evens, and Glenn E. Cunningham (latest). Dr. Arden Albee of the California Institute of Technology was the project scientist and Dr. Frank Palluconi of JPL the deputy project scientist.

3.2.2 Performance[15-37]

3.2.2.1 Spacecraft Performance

Table 5b shows the performance assessment of the spacecraft's key functions. The performance of the S/C will be considered during two phases of the mission. The first phase is the cruise phase between launch and the time of loss of contact, just before the initiation of propulsion system pressurization in preparation for Mars Orbit Insertion (MOI), approximately eleven months. During this time the S/C performed without any significant problems. The second phase commenced on 8/21/93 when the S/C was executing a sequence to pressurize the propulsion tanks three days later. As part of that sequence, the transmitter was turned off, and no signal was ever detected after that. The cause of the loss of contact can only be assigned in a probabilistic sense; there were no "smoking guns". Therefore, the performance of the MO subsystems during the eleven months of cruise can be discussed in terms of the mostly minor anomalies that occurred. The performance after the loss of contact is indeterminate. However, it is assumed that there was no performance thereafter because the Mars Balloon Relay beacon was not detected. The following paragraphs discuss what is known about the S/C, along with the most probable causes of the demise of the MO mission. The subsystem performance during cruise will be considered first.

Telecommunications

The only performance variation occurring with the telecommunications subsystem was the fact that the high-gain antenna (HGA) was misaligned by a 1¡. This caused a 6 dB loss of signal in the array normal spin (ANS) mode, which caused a reduction in downlink data rates in the outer cruise sequences. (The pointing error could have been corrected by commanding the gimbal actuators, but the project office decided that it was unnecessary.)

Although there were no problems with the flight hardware, there were a number of lost data packets due to problems with the DSN and MOSO ground data system (GDS). There was also a problem with corrupted telemetry with regard to Doppler data and range data used by the navigation team. There was a loss and/or corruption of the Mars Observer camera (MOC) image packets by the GDS.

Some of the loss of the MOC packets was caused by the connectivity to the MOSO H/W (ground hardware) Reed-Soloman Decoder-the problem was solved by converting to a new MOSO S/W Reed-Soloman Decoder. There were hardware problems at the DSN tracking stations and with GDS hardware provided by MOSO responsible for the reoccurring data outages. Data flow problems through DSN were fixed through delivery of new software and hardware. Most of the data was recovered by later playback of the recorded data.

Command and Data

Performance within the Command and Data Handling Subsystem was met by S/C hardware; however there was some data loss due to the ground hardware as noted above. The S/W problems included command timing bugs that required script logic changes to fix.

There were the expected occurrences of single-event-upsets (SEU) events commensurate with the point in time of the solar activity cycle. SEUs can also be caused by cosmic rays. The S/C automatically detected and corrected the effects of SEUs as planned in the design. They were not thought to be related to the MO loss-of-signal anomaly.

Attitude and Articulation

The performance of the Attitude and Articulation Control Subsystem hardware proceeded with only temporary malfunctions and no degradation. These temporary problems are discussed below. There were some problems with the flight software.

These included the loss of inertial reference caused by attitude disturbances fixed by modification of flight software in use for the star processing executive program (STAREX) script. The S/C entered Contingency Mode (solar panels are pointed toward the Sun, S/C rotates about the y-axis, the payload and the PDS are turned off to reduce electrical loads, on-board scripts are disabled, the HGA is disabled, and the LGA is enabled) after losing inertial reference as the result of a failed Sun-Monitor-Ephermeris-Check. The anomaly was related to a problem with star identification. Changes were made to the STAREX covariance matrix necessitated by bugs in the STAREX software. This problem required a number of fixes. These disturbances occurred during momentum desaturation events and when the reaction wheels' speed went through 0 (the result of static friction). These changes were expected to protect the S/C from reoccurrences of the inertial reference loss/contingency mode during the encounter and transition orbit period.

The hardware-caused problems included a temporary loss of inertial reference due to solar proton activity which was self-healing. A misaligned sun sensor head caused a slightly degraded sun sensor assembly performance during the sun coning mode, but the use of multiple heads kept performance in the nominal range. Also one of the sun sensor assembly heads was in the shadow of the solar array; this was corrected by use of a redundant sensor. The celestial sensor had a temporary loss of one fan due to solar protons causing slit pulsation. Redundancy allowed correction of the problem.

Electric Power

The only performance deviation within the power subsystem during cruise was the fact that the solar array operated at a higher temperature than predicted; however there was ample power margin to account for any deviations from the expected power output.

Propulsion

No problem was reported with the propulsion system during the cruise phase.

Thermal Control

The thermal control subsystem had a temporary malfunction in the temperature monitoring of the camera narrow angle component due to a failed thermistor. Alternate telemetry was available to correct the problem.

There were no other hardware problems, except that the S/C bus thermal model was updated, not an unusual situation. (Apparent discrepancies with the model and flight measurements were resolved when it was found that the power dissipation in the partial shunts was being computed incorrectly.)

Mechanisms

The full deployment of the high-gain antenna was delayed due possibly to a cold stiff cable and/or a hang-up on a bolt head. Since the deployment is a one-time only event, the delay had no significant impact on performance.

Other Subsystems

All other subsystems experienced mostly nominal performance during cruise. Possible catastrophic failures during propulsion system pressurization sequence are discussed below.

3.2.2.2 Payload Performance

Although no significant science was performed during cruise, except for the radio-science successful completion of the gravity wave experiment and the KABLE experiment, instruments were calibrated, and the imaging capability of the camera was tested. Therefore, this section will be concerned with information acquired during calibration of the instruments during cruise that indicated problems that would surface during the orbital phase of the mission.

Mars Observer Camera (MOC)

Astigmatism of the narrow-angle optics caused image distortion. This problem was mitigated by the application of about one Watt of energy to the mirror rim heater, and operated the camera at the point of best focus.

Pressure Modulator IR Radiometer (PMIRR)

The PMIRR had a loss of a channel-1 sideband prior to launch, but resources did not exist to fix the problem. It was felt that there was still enough capability to get enough useful data. Since this problem occurred before launch it was not recorded in Table 5b.

There was also a minor malfunction with the PMIRR resulting in operation below the required temperature after auxiliary power was turned off during a maneuver. This was fixed by permanently leaving on the auxiliary power and it should have had no impact on future operation of the instrument.

Magnetometer/Electron Reflectometer (MAG/ER)

The operation of the magnetometer was the one instance, up until the initiation of propulsion system pressurization, where the MO gave evidence of not being able to meet one of its requirements. The magnetometer suffered magnetic interference from the S/C caused by current loops resulting from the solar array operation. This compromised its ability to make dynamic (more serious of the two) and static magnetic field measurements of the Martian magnetic field. This could have been overcome during solar occultation (since the orbit around Mars was sun-synchronous, resulting in regular and predictable occultation periods of about 40 minutes per orbit) allowing substantial data to be taken at that time. Special calibration roll maneuvers were conducted during cruise to enable calibrating out the S/C residual field. Moreover, if the Martian magnetic field strength was at the upper limit of that expected it might have been possible to map it at other times as well. Therefore, the actual performance of the instrument would not have been known until the orbiting of Mars, although there was expectation of obtaining useful data.

The case of the electron reflectometer (ER) was shorted to the S/C chassis ground. This would result in the loss of the lowest-energy-range electrons. The ER data was supplemental to the magnetometer data, providing additional data from the electron trajectories. Although the ability of the ER to measure local surface magnetic fields was compromised, the supplemental nature of the ER measurements would cause only a minor impact on achieving mission objectives.

Other Instruments

There was no other indication of potential problems with the other instruments.

3.2.2.3 S/C Performance During Propulsion System Pressurization

As stated above, the performance of the S/C at the initiation of propulsion system pressurization prior to MOI can only be assessed in a probabilistic sense. This is done here by citing the most probable cause and other potential causes of loss of contact with the S/C as stated in the NASA Failure Review Board Final Report[15].

The most probable and two of the potential causes were related to the Propulsion and Pyro Subsystem. The most probable cause of loss of contact was catastrophic loss of control of the S/C by unintended mixing of NTO and MMH causing an explosive reaction. This could have been caused by leakage of the NTO through the check valves during the eleven-month cruise phase and mixing in the lines with the MMH during propulsion system pressurization.

A potential cause of the MO failure was a regulator failing to open, leading to bipropellant-tank overpressure and rupture, and destruction of the S/C. Another potential cause of the failure was a pyro valve failure causing the NASA Standard Initiator (NSI) to be expelled and puncturing some critical S/C component.

The other potential cause cited by the NASA Review Board was likely to have been located in the power subsystem. This was the shutdown of the power distribution to the S/C due to a power-supply-electronics power-diode insulation failure resulting in a short between the power bus and the chassis.

For more detail on these most probable and potential causes, and other potential failure modes, the reader is referred to the NASA Review Board Final Report[15] and the JPL Review Final Report[16].

3.2.2.4 Mission Objectives

Contact was lost with Mars Observer prior to the initiation of the primary mission. (The primary science objectives of the mission are shown in Table 7a). The only actual science accomplished was the successful completion of the gravity wave and KABLE experiments, which were not part of the primary mission. As stated above, one of the instruments, the magnetometer/electron reflectometer was degraded as a result of a S/C design deficiency. The extent of fulfillment of mission objectives by the instrument would not have been known until the Mars orbit phase. The other instruments had no known major problems at the time of loss of contact with the S/C.

Since there was no opportunity for MO to accomplish its primary mission objectives, the mission as a whole would be rated as a failure.

3.2.3 Summary of Results

Table 8 summarizes the spacecraft's performance by key functions. Table 9 and 10 give the number of anomalies and the various work-arounds that have been used to fulfill the mission requirements, with and without the payloads respectively. The overall performance rating for this S/C is shown in Tables 11 and 12, with the payloads included and excluded respectively. The degree to which the mission objectives were accomplished is shown in Table 13. Table 14 shows examples of several possible correlations to functional performance and anomalies.

3.2.4 Conclusions

Except for not providing a magnetically clean environment for the magnetometer experiment, Mars Observer met all of its requirements up to the point of pressurization of the bipropellant propulsion subsystem prior to orbit insertion. There were the usual, expected ground support problems in the check-out phase of a flight mission. These did not significantly impact the cruise phase of the mission, nor were they expected to impact the ability of the S/C to carry out its primary mission. This is all that can be said with certainty about Mars Observer. The actual cause of the failure of the mission and the actual identity of the culprit subsystem or subsystems could not be deduced; only the most probable cause or causes of the mission failure within the time period of loss of contact with the S/C could be investigated.

References

1. Farguson, C., "Flight Performance Characterization (FPC) RTOP: NASA Unmanned Spacecraft/Instrument Performance Report #1", JPL Doc. No. D-11619, September 14, 1994.

2. TOPEX/POSEIDON Project - Mission Description, JPL Doc. No. D-601, Rev. B, December, 1985.

3. TOPEX/POSEIDON Project - Mission and Systems Requirements, JPL Doc. No. D-5901, April, 1989.

4. TOPEX/POSEIDON Project - Satellite System Design Specification, Fairchild Space Doc. No. 968-PF1000, Rev. H, May, 1990.

5. TOPEX/POSEIDON Project - Satellite System Performance Requirements, JPL Doc. No. D-2226, Rev. D, January, 1990.

6. TOPEX/POSEIDON - A United States/France Mission, JPL Doc. No. D-9953, August, 1992.

7. TOPEX/POSEIDON Fact Sheet, Office of Public Information, JPL, July 18, 1991.

8. TOPEX/POSEIDON Project - Satellite/Sensors Performance Characteristics Workshop #1, JPL Doc. No. D-10596, November 15, 1992.

9. TOPEX/POSEIDON Project - Satellite/Sensors Performance Characteristics Workshop #2, September 15, 1993.

10. TOPEX/POSEIDON Project - Satellite/Sensors Performance Characteristics Workshop #3, July 26, 1994.

11. Problem/Failure Reports (PFRs), Problem/Failure Operations (PFO) Center, JPL.

12. Mars Observer Project - Mission Requirements Document, JPL Doc. No. D-2940, Rev. c, December 1990.

13. Mars Observer Project - Spacecraft Requirements, JPL Doc. No. D-1704, Rev. A, March 24, 1989.

14. Mars Observer Project Fact Sheet, Office of Public Information, JPL, July 23, 1992.

15. "Report of the Mars Observer Mission Failure Investigation Board", Vol. 1, NASA, December 31, 1993.

16. "Mars Observer Loss of Signal: Special Review Board Final Report", JPL Publication 93-28, November, 1993.

17. (For Magnetometer/ER problems) private communication with Deputy Project Scientist, Frank Palluconi.

18. Mars Observer Project Management Report, Levels 2,3 26 October 1992.

19. Mars Observer Project Management Report, Levels 2,3 17,18 Dec. 1992.

20. Mars Observer Project Management Report, Levels 2,3 22 January 1993.

21. Mars Observer Project Management Report, Levels 2,3 26 March 1993.

22. Mars Observer Project Management Report, Levels 2,3 23 April 1993.

23. Mars Observer Project Management Report, Level 3, 27 May 1993.

24. Mars Observer Project Management Report, Levels 2,3 24,25 June 1993.

25. Mars Observer Project Management Report, Levels 2,3 21,23 July 1993.

26. Mars Observer NASA Quarterly Review, 11/18/92.

27. Mars Observer NASA Quarterly Review, 2/22/93.

28. Mars Observer Spacecraft Anomaly Summary, 11/13/92.

29. Mars Observer Spacecraft Anomaly Summary, 2/17/93.

30. Mars Observer ISA Log and PFO Center PFR Database.

31. Brejcha, A.G., "Office of Engineering and Review (OER) Report for September/October 1992", JPL IOM 52AGB 92-116, Revised 11/13/92.

32. Brejcha, A.G., "Office of Engineering and Review (OER) Report for November/December 1992", JPL IOM 52AGB 93-003, 1/12/93.

33. Brejcha, A.G., "Office of Engineering and Review (OER) Report for January/February 1993", JPL IOM 52AGB 93-016, 3/04/93.

34. Brejcha, A.G., "Office of Engineering and Review (OER) Report for March/April 1993", JPL IOM 52AGB 93-037, 5/12/93.

35. Brejcha, A.G., "Office of Engineering and Review (OER) Report for May/June 1993", JPL IOM 52AGB 93-043, Revised 6/29/93.

36. Brejcha, A.G., "Office of Engineering and Review (OER) Report for July/August 1993", JPL IOM 52AGB 93-066, 9/07/93.

TABLE 1: S/C SYSTEMS: Mission Characteristics

Characteristic	TOPEX/POSEIDON	MARS OBSERVER	
Launch Date(S)	8/10/92	9/25/92	
Launch System	Ariane 42P, 3-stage	Titan III with Transfer Orbit Stage	
Mission Duration, Yr. Design, Total Cruise Tour	 5 yrs 0 5 yrs	 3+ yrs 11 months 2+ yrs	
Mission Termination Date	1998 3 yr primary 3 yr extended	Nov. 95 (with options) (original planned date) Aug. 93 (actual date)	
Mission Destination	Earth orbit	Mars	
Mission Type	Earth satellite	Interplanetary, inner planet, orbiter	
Solar AU Range	1	1.53	
Orbit Description	Circular	Circular, polar, sun-synchronous	
Expected Significant Environments:			
Radiation	Geomagnetically trapped particles; solar flare galactic cosmic ray	Solar and cosmic particles	
Thermal	Solar radiation, albedo, Earth IR	Space, solar radiation, Mars albedo	
Thermal Cycling?	yes, Earth shadow of sun	yes, Mars shadow of sun	
Micrometeroid	Man-made space debris environment	Not as severe as outer planets	
Dynamic	launch, microphonics, pyroshock	launch, microphonics, pyroshock	

TABLE 2: S/C SYSTEMS: Program Characteristics

Characteristic	TOPEX/POSEIDON	MARS OBSERVER	
Program Management NASA Center	JPL (for NASA) French Space Agency (CNES)	JPL (for NASA)	
Prime Contractor	S/C bus - Fairchild Space Co.	General Electric Astro-Space Div., Princeton, NJ.	
Other Major Contractors	Poseidon altimeter - Alcatel Espace; Antenna pointing system, reaction wheel assemblies - Honeywell Satellite Systems Div.; DORIS receiver - Dassault Electronique	Titan III launch vehicle - Martin Marietta Commercial Titan, Inc. Denver, CO. Transfer Orbit Stage - Orbital Sciences Corporation, Vienna, VA.	
Type of contract	Cost plus	Fixed price, plus on-orbit performance fee	
#**Costs (Per Mission) (Fy94$M)**			
Pre-project	33.6	19.0	
Project Mgt.	10.6	21.7	
Spacecraft Develop.	266.9	274.2	
Science	96.0	223.9	
Other	18.6 (GPSDR)	-	
Mission Ops(Pre-Launch)	11.0		
Total Develop. (launch +30 days)	464.6	571.8	
Mission Ops & Data Analysis (Post-Launch)*	130.5 (thru FY 98)	116.0 (budgeted thru FY 96)	
Total Project Costs	595.1 (thru FY 98)	687.8 (budgeted thru FY 96, not totally expended due to early termination)	
# Of S/C	1	1	
Complexity Rating	TBD	TBD	
Class(NMI 8010.1) S/C	B	A	

Class(NMI 8010.1) Payload	B (except for CNES altimeter)	B	
Product Assurance Program Control Standard (JPL D-1489, MILSTD1540, SPAR GEVS)	D-1489 (with exceptions)	D-1489 (with exceptions)	

#Cost numbers supplied from JPL System Analysis Section, 311; In FY 94 dollars.
*Unless otherwise noted post-launch costs assume no extended mission.

TABLE 3: S/C SYSTEMS: Spacecraft Characteristics

Characteristic	TOPEX/POSEIDON	MARS OBSERVER	
Mass, kg			
Eng. subsys.	1832.7	1071	
Science/payload	430 (sensors)	156	
Total dry	2262.7	1227	
Propellants	217.3	1346	
Total injected	2480	2573	
Other	Ariane launch vehicle: 4000 Kg	Titan III launch vehicle:	
No. Of Parts	100,000 (assembly level)	61,000	
Power, W			
Source	Solar array (1)	Solar array (1)	
Voltages (Bus)	28 V reg. dc	28 V reg. dc	
Capability BOL	3380 W	1130 W	
Capability EOM	2140 W (after 5 years)		
Storage	Three 50AHr Ni-Cd batteries (each with 22 cells in series)	Two 42AHr Ni-Cd batteries	
Instrument Payload			
total # instruments	6	7	
total # investigations	38 investigation teams (200 scientists, 11 countries)	13 investigations	
field & particles	-	MAG/ER	
imaging	2 altimeters	MOC	
spectro-radiometric	Microwave radiometer	PMIRR,TES, GRS	
others	Laser retroreflector; DORIS system receiver; GPS demonstration receiver	RS	
Telecom			
High Gain			
type	Parabolic dish: TDRS type, deployable	Parabolic articulating: (deployed on a 6-m boom)	
dia, m	1.2	1.45	
articulating?	no	yes	
freq. band (S,X,Ka, Ku)	S (HGA) Ku , C (NASA altimeter)	X, Ka	

max. power output	20 W (Ku) 20 W (C)	22 W (44 W total downlink RF power, including line loss from TWTA to antenna)	
max downlink, kbps	512	85.3 (downlink max) 0.5 (uplink max)	
Other Antennas	Zenith omni antenna Nadir omni antenna	3 low-gain antennas	

TABLE 3: S/C SYSTEMS: Spacecraft Characteristics (cont.')

Characteristic	TOPEX/POSEIDON	MARS OBSERVER	
Command & Data	Command and Data Handling Subsystem (CDHS) (Comprised of satellite central unit, multiplex data bus, bus coupler units, and remote interface/expander units)	Command and Data Handling Subsystem (CDHS) with Payload Data Subsystem	
Processor	80C86 processor	On-board microprocessors: 1750A's/80C86	
Memory capacity. (RAM)	65536 18-bit words	16-bit words SCP: 96000 bytes EDF: 22000 bytes PDS: 64000 bytes	
Auto. Fault Correction?	yes	yes	
Tape recorder storage capacity., Mbit.	3 recorders >500 Mbit each	138 Mbit total	
Attitude & Articulation			
Processor	80C86 processor	On-board microprocessors: 1750A's (Function done by CDHS processors.)	
Memory capacity. (RAM)	65536 18-bit words	n/a	
stabilization	3-Axis	3-Axis	
control	Thrusters (16 total, 4@22N, 12@1N); Reaction wheel; Magnetoquers; Gyros	Thrusters (24 total, 4@490N, 4@22N, 8@4.5N, 8@0.9N); Reaction wheels;	
sensors	2 axis star tracker; 2 axis fine sun sensor; Earth sensor; 3 axis magnetometer; Course sun sensors and sun presence sensors; 3 axis gyros	Horizon sensor; Celestial Sensor Assembly; Gyros, 3 axis, 2 DOF; Sun sensors; Accelerometers (4)	
Articulation (# of elements)	1	4	
types	Solar array	Solar array - 2 axis HGA - 2 axis	
pointing accuracy, mrad			

bus	2.44	Pointing accuracy: Control: 10 Knowledge: 3	
scan platform	Control: 2.44 Knowledge: 1.22		
others	Solar array sun pointing HGA pointing to TDRS	Solar array pointing HGA pointing - 2 axis	

TABLE 3: S/C SYSTEMS: Spacecraft Characteristics (cont.')

Characteristic	TOPEX/POSEIDON	MARS OBSERVER	
Mechanisms			
# of deployables	6	4	
type	Ariane L/V 1 solar panel array High gain antenna boom GPS antenna Zenith omni antenna Nadir omni antenna	1 solar panel array High gain antenna boom 2 instrument booms	
Propulsion			
total delta-v, m/s	>160	2249 (through mapping orbit insertion) 45 (mapping orbit maneuver)	
Engine type and Propellants:			
OI	Ariane 42P 3-stage Jettisonable Liquid monopropellants with 2 solid propulsion strap-on boosters	Titan III (liquid propellants with 2 solid propulsion strap-on boosters) with Transfer Orbit Stage (solid propellant)	
ACS/TCM	Hydrazine thrusters: 4 @22 N; 12 @1 N	Bi-propellant (MMH/NTO): 4 @490N 4 @22N; Hydrazine thrusters: 4 @4.5N (orbit trim) 4 @0.9N (momentum unloading and steering)	
Thermal			
elements	Multilayer blankets, louvers, heaters, radiators, heatpipes	Multilayer blankets, louvers, heaters, radiators	

TABLE 4a: INSTRUMENT SYSTEMS: Program Characteristics

TOPEX/POSEIDON	**Principal Investigator, affiliation**	**Where designed/built**	**product assurance program control standard** (JPL D-1489, MILSTD1540, SPAR GEVS)	**Class(NMI 8010.1)**
Dual frequency radar altimeter - (ALT)	Managed by Goddard Space Flight Center	Johns Hopkins University's Applied Physics Laboratory	JPL D-1489 (with exceptions)	B
TOPEX microwave radiometer - (TMR)	Managed by JPL	JPL	JPL D-1489 (with exceptions)	B
Laser retroreflector array - (LRA)	Manage by Goddard Space Flight Center	Johns Hopkins University's Applied Physics Laboratory	JPL D-1489 (with exceptions)	B
DORIS dual Doppler tracking system receiver - (DORIS)	Managed by CNES.	Dassault Electronique (receiver); CEIS Espace (beacons); CEPE and OSA (quartz oscillators).	JPL D-1489 (with exceptions)	n/a
Single-frequency solid-state radar altimeter - (SSALT)	Managed by CNES.	Designed by CNES. Built by Alcatel Espace, France.	JPL D-1489 (with exceptions)	n/a
GPS demonstration receiver - (GPSDR)	Managed by JPL	Motorola	JPL D-1489 (with exceptions)	B

TABLE 4b: INSTRUMENT SYSTEMS: Program Characteristics

MARS OBSERVER	**Principal Investigator, affiliation**	**Where designed/built**	**product assurance program control standard** (JPL D-1489, MILSTD1540, SPAR GEVS)	**Class(NMI 8010.1)**
Gamma Ray Spectrometer	William Boynton (team leader), University of Arizona.	Martin Marietta Astronautics Group	JPL D-1489 (with exceptions)	B
Thermal Emission Spectrometer	Philip R. Christensen, Arizona State University.	The Santa Barbara Research Center (instrument contractor).	JPL D-1489 (with exceptions)	B
Mars Observer Camera	Michael C. Malin, Malin Space Science Systems, Inc.	Caltech	JPL D-1489 (with exceptions)	B
Mars Observer Laser Altimeter	David E. Smith, NASA's Goddard Space Flight Center.	Goddard Space Flight Center	JPL D-1489 (with exceptions)	B
Pressure Modulator Infrared Radiometer	Daniel J. McCleese, JPL.	JPL	JPL D-1489 (with exceptions)	B
Radio Science	Leonard Tyler, (team leader), Stanford University.	uses spacecraft's radio augmented by an ultrastable oscillator built by Applied Physics Lab of John's Hopkins University	JPL D-1489 (with exceptions)	B
Magnetometer/ Electron Reflectometer	Mario H. Acuna, Goddard Space Flight Center	The magnetometer was built by GSFC and the electron reflectometer by the French Centre National d'Etudes Spatiales.	JPL D-1489 (with exceptions)	B

TABLE 6a- INSTRUMENT SYSTEMS: PERFORMANCE ASSESSMENT

TOPEX/POSEIDON	Functional perf. rating	Lifetime (% of design)	Science Objectives Accomplished
Dual frequency radar altimeter - (ALT)	1 (to date)*	TBD (2.3 years to date)	Accomplished (to date).
TOPEX microwave radiometer - (TMR)	1 (to date)*	TBD (2.3 years to date)	Accomplished (to date).
Laser retroreflector array - (LRA)	1 (to date)*	TBD (2.3 years to date)	Accomplished (to date).
DORIS dual Doppler tracking system receiver - (DORIS)	1 (to date)*	TBD (2.3 years to date)	Accomplished (to date).
Single-frequency solid-state radar altimeter - (SSALT)	1 (to date)*	TBD (2.3 years to date)	Accomplished (to date).
GPS demonstration receiver - (GPSDR)	1 (to date)*	TBD (2.3 years to date)	Accomplished (to date).

Performance Rating: **1**: Yes, no significant anomalies; **2**: Yes, with work-around of significant anomaly (incl. redundancy); **3**: Partial, minor degradation ; **4**: Partial, significant degradation or persistent malfunctions; **5**: Performance not met. Work-*around:* **N**=none; **O**=operational; **S**=software fix; **R**=redundancy.
PM = through primary mission, **EM** = through extended mission
* Based on returned data so far.

TABLE 6b- INSTRUMENT SYSTEMS: PERFORMANCE ASSESSMENT

MARS OBSERVER	Functional perf. rating	Lifetime (% of design)	Science Objectives Accomplished
Gamma Ray Spectrometer (GRS)	1	n/a	Instrument turned on for a long time during cruise and was operational. Mission aborted prior to measurements on Martian environments.
Thermal Emission Spectrometer (TES)	1	n/a	Instrument turned on some time during cruise and was operational. Mission aborted prior to measurements on Martian environments.
Mars Observer Camera (MOC)	2 (some images were taken)	n/a	Instrument turned on for a long time during cruise and was operational. Mission aborted prior to measurements on Martian environments.
Mars Observer Laser Altimeter (MOLA)	1	n/a	Instrument turned on some time during cruise and was operational. Mission aborted prior to measurements on Martian environments.
Pressure Modulator Infrared Radiometer (PMIRR)	2 (instrument turned on)	n/a	Instrument turned on some time during cruise and was operational. Mission aborted prior to measurements on Martian environments.
Radio Science (RS)	1	n/a	Instrument turned on for a long time during cruise and was operational. Mission aborted prior to measurements on Martian environments, but radio science participation resulted in successful completion of gravity wave experiment.
Magnetometer/ Electron Reflectometer (MAG/ER)	4 3	n/a	Instrument turned on some time during cruise and was operational. Mission aborted prior to measurements on Martian environments, potential for magnetic data to be taken while in solar occultation, ER data supplemental to magnetometer data.
Mars Balloon Relay	1	n/a	Mission aborted prior to measurements on Martian environments.

Performance Rating: **1**: Yes, no significant anomalies; **2**: Yes, with work-around of significant anomaly (incl. redundancy); **3**: Partial, minor degradation ; **4**: Partial, significant degradation or persistent malfunctions;
5: Performance not met. Work-*around:* **N**=none; **O**=operational; **S**=software fix; **R**=redundancy.
PM = through primary mission, **EM** = through extended mission
* Based on returned data so far.

TABLE 8: S/C PERFORMANCE COMPARISON BY KEY FUNCTION

Engineering Function	TOPEX perf. rating	TOPEX mission impact	MO - CRUISE perf. rating	MO - CRUISE mission impact	MO-Prior to MOI perf. rating	MO-Prior to MOI mission impact	perf. rating	mission impact
TELECOM								
Except during known or specified time intervals, provide a two-way communications link between the spacecraft and the DSN.	2 [M]	negligible	2 [0]	negligible	indeter-minate			
For commands, provide for an uplink at S-band.	1		n/a		n/a			
For commands, provide for an uplink at X-band.	n/a		1		indeter-minate			
For telemetry, provide for a downlink at X-band	n/a		1		indeter-minate			
For telemetry, provide for a downlink at S-band	1		n/a		n/a			
If applicable, provide for receiving relay data from another vehicle or source, at rates and bands specified.	1		n/a		n/a			
COMMAND AND DATA								
Decode, store, execute and distribute commands.	2 [Ox2] 3 [R] [M]	minor so far	1		indeter-minate			
Provide for S/C clock.	1		1		indeter-minate			
Acquire and format all engineering and science data.	1		1		indeter-minate			
Record and play back data at commanded times and rates.	2 [M]	none	1		indeter-minate			
ATTITUDE /ARTICULATION								
Provide S/C stabilization, orientation and pointing.	2 [Ox3] [H] [R]	minor	2[H]2[M] 2[Rx2]	negligible	indeter-minate			
Control major and minor propulsive maneuvers.	1 [M]	none	1		indeter-minate			
Articulation: solar panel	1		n/a		indeter-minate			
Articulation: science platform	n/a		n/a		n/a			
Articulation: relay antenna	n/a		n/a		n/a			
Articulation: spin bearing ssy	n/a		n/a		n/a			
PROPULSION								
Provide thrust for orbit insertion.	1		n/a		5 [x3] (see Tab. 5b)	catastro-phic		
Provide thrust for trajectory control maneuvers	1		1		n/a			
Provide thrust for attitude control	1		1		n/a			
POWER								

Provide for Energy Conversion	1		1		5 (see Tab. 5b)	catastro-phic		
Provide for energy storage	1		1		indeter-minate			
Provide for power control and regulation	1		1		indeter-minate			
Provide for firing of pyro squibs.	1		1		indeter-minate			
Condition and regulate power for probe checkout	n/a		1		indeter-minate			

*Performance: **1**: Yes, no significant anomalies; **2**: Yes, with recoverable temporary loss or malfunction; **3**: Partial, minor degradation; **4**: Partial, significant degradation; **5**: Performance not met.
Type of fix: **R**=redundancy, **O**=operational or software, **M**=resource margin , **H**=self healing
PM = primary mission; **EM** = extended mission

TABLE 8: S/C PERFORMANCE COMPARISON BY KEY FUNCTION (CONT.')

	TOPEX		MO - CRUISE		MO-Prior to MOI			
Engineering Function	*perf. rating*	*mission impact*	*perf. rating*	*mission impact*	*perf. rating*	*mission impact*	*perf. rating*	*mission impact*
MECHANISMS								
Provide S/C separation from L/V or TOS	1		1		indeter-minate			
Provide S/C separation from prop. module	1		1		indeter-minate			
Provide for probe separation	n/a		n/a		n/a			
Provide for deployment and/or latching of: solar panels or RTG booms	1		1		indeter-minate			
Provide for deployment and/or latching of: scan /science platform	n/a		n/a		n/a			
Provide for deployment and/or latching of: MAG boom	n/a		1		indeter-minate			
Provide for deployment and/or latching of: LGA 2 boom	n/a		n/a		n/a			
Provide for deployment and/or latching of: High Gain Antenna	n/a		2[M]	negligible	indeter-minate			
THERMAL								
Control temperature of S/C assemblies to within specified limits. Aid in micro-meteoroid protection.	2[Ox2], [M]	negligible	2 [R] 2 [M] 2 [O]	negligible minor negligible	indeter-minate			
STRUCTURE								
Provide mechanical support and alignment; radiation and micrometeorite protection; vibration protection; etc.	1		1		indeter-minate			
CABLING (interfaces)								
Provide electrical interconnections, incl. shielding, EMI protection, and grounding.	1		1		indeter-minate			
PAYLOAD	*PM:* 2(Mx2) 1 1 1[O] 1[O], 2[M] 1	minor minor	2[Ox2] 4 [O] 3 [m] 1 1 1 1 1	negligible-minor signifi-cant minor	indeter-minate			

*Performance: **1**: Yes, no significant anomalies; **2**: Yes, with recoverable temporary loss or malfunction; **3**: Partial, minor degradation; **4**: Partial, significant degradation; **5**: Performance not met.

Type of fix: **R**=redundancy, **O**=operational or software, **M**=resource margin , **H**=self healing

PM = primary mission; **EM** = extended mission

TABLE 9: SUMMARY OF KEY FUNCTIONAL PERFORMANCE , ANOMALIES, AND WORK-AROUNDS (not incl. payload)

	TOPEX		MO - CRUISE		MO-Prior to MOI			
PERFORMANCE	**PM** (2.3 yrs to date)	**EM (n/a)** (2 yrs planned)	**PM**	**EM**	**PM**	**EM**	**PM**	**EM**
# of key functions met w/o significant anomalies [1]	19 (76%)		34 (79%)		indeter-minate			
# with recoverable temporary loss [2]	5 (20%)		9 (11%)		indeter-minate			
# with minor degradation [3]	1 (4%)		0		indeter-minate			
# with significant degradation [4]	0		0		indeter-minate			
# not met [5]	0		0		^{3}1			
RELEVANT ANOMALIES Total**:**	15		9		^{3}1			
Ave. per year	~7.0		~9		n/a			
WORK-AROUNDS Total**:**	14		9		n/a			
Redundancy	2		3		n/a			
Operational/ software	7		2		n/a			
Resource margin	4		2		n/a			
Self healing	1		2		n/a			

PM = through primary mission, **EM** = through extended mission

TABLE 10: SUMMARY OF KEY FUNCTIONAL PERFORMANCE , ANOMALIES, AND WORK-AROUNDS (incl. payload)

	TOPEX		MO - CRUISE		MO-Prior to MOI			
PERFORMANCE	**PM** (2.3 yrs to date)	**EM (n/a)** (2 yrs planned)	**PM**	**EM**	**PM**	**EM**	**PM**	**EM**
# of key functions met w/o significant anomalies [1]	24 (75%)		37 (74%)		indeter-minate			
# with recoverable temporary loss [2]	7 (22%)		11 (22%)		indeter-minate			
# with minor degradation [3]	1 (3%)		1 (2%)		indeter-minate			
# with significant degradation [4]	0		1 (2%)		indeter-minate			
# not met [5]	0		0		≥1			
RELEVANT ANOMALIES Total**:**	17		13		≥1			
Ave. per year	~8.5		~13		n/a			
WORK-AROUNDS Total**:**	17		13		n/a			
Redundancy	2		3		n/a			
Operational/ software	7		5		n/a			
Resource margin	7		3		n/a			
Self healing	1		2		n/a			

PM = through primary mission, **EM** = through extended mission

TABLE 11: OVERALL SPACECRAFT PERFORMANCE RATING* (not incl. payload performance)

	TOPEX	MO - CRUISE	MO-Prior to MOI	
RATING: Primary Mission (PM) Extended Mission (EM) (R=redundancy)	1 (to date) (R=2) (S/C supported the mission, but its capability is slightly diminished due to hardware failures)	1	5	
Lifetime	TBD (2.3 yrs to date)	11 months	n/a	
EXPLANATION	Two-thirds of the mission completed. 3 hardware failures and many SEUs, but S/C continues to support mission.	See Table 5b for S/C performance during cruise.	Catastrophic failure after 11 months of cruise at the start of propulsion system pressurization.	

* **1**: S/C fully supports mission.
2: S/C capability to support mission is slightly diminished.
3: S/C capability to support mission is moderately diminished.
4: S/C capability to support mission is significantly diminished.
5: S/C failure

TABLE 12: OVERALL SPACECRAFT PERFORMANCE RATING* (incl. payload performance)

	TOPEX	MO - CRUISE	MO-Prior to MOI	
RATING: Primary Mission (PM) Extended Mission (EM) (R=redundancy)	1 (to date) (R=2) (The instruments are working well, only the S/C capability is slightly diminished due to hardware failures)	1	5	
Lifetime	TBD (2.3 yrs to date)	11 months	n/a	
EXPLANATION	S/C: See Table 4 Payload: Fully supports mission to date.	See Table 6b for payload calibration and performance during cruise.	Catastrophic failure of S/C after 11 months of cruise at the start of propulsion system pressurization.	

* **1**: S/C fully supports mission.

2: S/C capability to support mission is slightly diminished.
3: S/C capability to support mission is moderately diminished.
4: S/C capability to support mission is significantly diminished.
5: S/C failure

TABLE 13: MISSION OBJECTIVES ACCOMPLISHED

	TOPEX	**MO - CRUISE**	**MO-Prior to MOI**
Accomplished?	Fully (to date)	Yes	No
Explanation	Accomplished so far. Had 3 hardware failures, but did not impact mission objectives.	No significant problems during 11 months of cruise - successful completion of gravity wave and KABLE experiments.	Loss of contact with S/C after initiation of propulsion system pressurization prohibited accomplishment of mission objectives.

TABLE 14: S/C PERFORMANCE COMPARISONS WITH OTHER CHARACTERISTICS

	TOPEX	**MO - CRUISE**	**MO-Prior to MOI**	
S/C Perf. Rating (w/o payload)	1 (to date) (R=2)	1	5	
Relevant anomalies per year of primary mission. (w/o payload)	~7.0	9	1 or more	
S/C Perf. Rating (with payload)	1 (to date) (R=2)	1	5	
Relevant anomalies per year of primary mission. (with payload)	~8.5	13	n/a	
Class	B	S/C A, payload B		
Mgmt. center/ Implementation mode	JPL/ System contract: S/C bus - Fairchild Space Co.;	JPL/System contract: General Electric Astro-Space Div. (now Martin Marietta Corp.)		
Type of contract	cost plus	Fixed price, plus on-orbit performance fee		
S/C complexity	Medium	Medium		
Mission complexity	(not determined for this task)	(Not determined for this task)		
Product Assurance Program	JPL D-1489 equivalent (with exceptions)	JPL D-1489 equivalent (with exceptions)		
Years of Operation Primary mission: Extended mission:	 2.3 yrs (to date) 	 11 months n/a		

Severe Mission Environments	**Thermal:** Solar, earth IR, albedo **Radiation**: High altitude Earth atmosphere	**Thermal**: Solar, albedo **Radiation**: Solar protons, cosmic rays		

OBJECTIVE	MET? (total, partial, not)	COMMENTS
LIFETIME:		
The mission is designed for a 3 year life, with enough margin on expendable resources for extension to 5 years.	TBD	2.3 years to date (Nov.94).
% OF LIFETIME MET	TBD	
SCIENCE OBJECTIVES:		
Measure (determine) sea level (geocentric), with each ground track repeat cycle, to an accuracy of ±14 cm (1 sigma) under typical ocean conditions, with small geographically correlated errors.	TBD	Objectives met to date.
Measure (determine) sea level distance from the satellite, when averaged along-track over a distance of 20 km, to a precision of ±2.4 cm (1 sigma) under typical ocean conditions.	TBD	Objectives met to date.
Measure (determine) sea level at least every 20 km along a grid of subsatellite tracks to minimize the spatial aliases of small-scale sea level variability.	TBD	Objectives met to date.
Measure (determine) sea level along a grid of subsatellite tracks fixed to the earth to minimize the influence of the geoid on measurements of time-variable topography.	TBD	Objectives met to date.
Measure (determine) sea level in such a manner that tidal signals shall not be aliased into semiannual, annual, or zero frequencies, or into frequencies close to these.	TBD	Objectives met to date.
Measure (determine) sea level along a grid of subsatellite tracks that make it possible to determine two orthogonal components of surface slope with comparable accuracy.	TBD	Objectives met to date.
Measure (determine) sea level at least as far south as the southern limit of the Drake Passage (62 deg).	TBD	Objectives met to date.

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effect On Individual Investigations:
TELECOM								
Except during known or specified time intervals, provide a two-way communications link between the spacecraft and the DSN.	2[M]	Minor	Occasional disruptions, due to NASCOM data line problems					
Support command and telemetry through TDRSS Single- and Multi-Access channel.	1		none					
For commands, provide for an uplink at any specified data rate, at X-band.	n/a							
For commands, provide for an uplink at any specified data rate at S-band.	1		none					
For telemetry, provide for a downlink at specified rates at X-band.	n/a							
For telemetry, provide for a downlink at specified rates at S-band.	1		none					

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effect On Individual Investigations:
COMMAND AND DATA								
Decode, store, execute and distribute commands	2[O]	Temporary malfunctions	Status buffer message not stored and received through downlink telemetry due to error in on-board computer flight software not setting flags correctly.	8/18/92 (59027) ISA 3163	Telemetry knowledge	*Software patch*	Minor, temporary loss of telemetry.	None
	2[O]	Temporary malfunctions	Telemetry monitor turned off NASA altimeter due to load shedding and battery discharge.	8/30/92 (59029) ISA 3340	Telemetry monitoring	*Software patch*	none	Minor, temporary loss of function of altimeter.
	3[R]	Temporary loss	Remote interface unit passive analog RIU-6A (PA) telemetry failed due to electrostatic discharge- induced failure of its multiplexer.	6/4/94 (59033) ISA 10114	Telemetry data and monitoring	Use redundant telemetry unit RIU-6B.	Loss of redundancy on remote interface unit.	None
	3[M]	Loss of the remote interface RIU-6B expander unit channels information.	Expander unit on the B-side of RIU-6B (E/U) failed due to ESD, similar to RIU-6A failure.	7/6/94 (59034) ISA 10096	Telemetry data and monitoring	Remain on RIU-6B. Read high gain antenna gimbal temperatures only.	Loss of data on many temperature measurements .	Minor, so far.
Provide S/C clock.	1		none					
Acquire and format all engineering and science data	1		none					

Record and play back data at command times and rates.	2[M]	minor	Sporadic changes in AC excursions & shift in DC characteristics on all tape recorders.		Data play back	None	Negligible	

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effect On Individual Investigations:
ATTITUDE AND ARTICULATION								
Provide 3-axis S/C stabilization, orientation and pointing.	2[O]	Temporary loss	Flight software not performing gyro corrections properly.	8/14/92 (59026) ISA 3675	Attitude knowledge	*Software patch* Loading correct scale factors.	Minor, temporary loss of accuracy of data.	None
	2[H]	Temporary loss (no permanent damage)	Spikes in earth sensor assembly position data, saturated fine telemetry channel - caused by SEUs over South Atlantic Anomaly.	8/19/92 (59028) ISA 3164	Attitude knowledge	Fault detection and control limits relaxed.	Minor, temporary loss of attitude knowledge.	None
	2[O]	Temporary loss	NASA altimeter sees a pointing error of ~0.6°.	10/23/92 (59030) ISA 4210	Altimeter pointing accuracy	*Software patch* and re-calibration to get pointing within specification.	Minor, temporary loss of accuracy of data.	Temporary loss of pointing accuracy of instrument.
	2[R]	Temporary loss	Advanced star tracker (ASTRA1-B) failed to acquire stars.	11/30/92 (59031) ISA 4219	Attitude knowledge	Use redundant star tracker ASTRA1-A.	Minor, temporary loss of accuracy of data.	None
	2[O]	Temporary loss	Digital fine sun sensor experienced false sun presence readings due to glint reflections within its field of view	12/30/92 (59032) ISA 2858	Attitude tracking	Updating flight software	Minor, temporary loss of attitude tracking.	None

Control major and minor propulsive maneuvers.	1		none					
Articulate 1 DOF solar panel to platform with accuracy of 35 mrad.	1[M]	none	Solar array resonance		Gyro gitter	None	Negligible	

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effect On Individual Investigations:
PROPULSION								
Provide thrust for payload insertion into earth orbit (Ariane 42p).	1		none					
Provide thrust for trajectory control maneuvers: ²1.5% error or ²0.4 mm/s for 1-N thrusters and ²16 mm/s for 22-N thrusters.	1		none					
Provide thrust for attitude control maneuvers.	1		none					
POWER/PYRO								
Provide for energy conversion; solar panel	1		none					
Provide for power control, regulation and distribution	1		none					
Provide for the firing of pyro squibs.	1		none					
Provide for energy storage: rechargeable batteries	1							
MECHANISMS								
Provide for solar panel deployment	1		none					
Provide for separation from Solid Rocket Motor	1		none					
Provide for S/C separation from launch vehicle.	1		none					
Provide for HGA deployment	1		none					

Provide for GPS antenna deployment	1		none					

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effect On Individual Investigations:
THERMAL								
Maintain control of the temperature of all S/C engineering and science subsystems and their elements/components within specified limits. Aid in providing micro meteorite protection.	2[O], [M] 2[O]	During record and playback cycles, the temperature of the tape recorders TR-A and TR-C electronic unit higher than limit The HGA gimbal operating temperature slightly higher than expected	53.44 ¡C actual, 50 ¡C required limit Characteristic of the solar exposed surfaces of the Y gimbal changed due to portions of the silverized Teflon coming off, resulting in a surface with a higher absorption to emission ratio.		Tape recorders data record and playback. High gain antenna	The high limit raised to 55 ¡C and the recorder operation procedure was changed. The operating temperature of the gimbal is now limited by maintaining the Y gimbal in the negative position for a long time.	None None	None None
STRUCTURE								
Provide mechanical support and alignment; provide radiation and micrometeroid protection; protect equipment from excessive vibration, etc; aid in assuring an equipotential surface.	1		none					
CABLING								

Provide the electrical interconnections between all S/C engineering and science subsystems, between subsystem elements, and those needed for interfacing with a separable vehicle system, with the L/V, with jettisonable elements, and with the launch complex. Provide for appropriate shielding, EMI protection, and grounding	1		none						

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effect On Individual Investigations:
PAYLOAD								
Dual frequency radar altimeter (ALT)- measures height to precision 2.4 cm altitude.	2[M]	1-2% data loss	Numerous SEUs		Detector	Added alarms at the project operations control center to alert science data hang-up and developed a new streamlined procedure for faster ground recovery from SEUs to minimize data loss.	Minimal	
	2[M]	Traveling wave tube amplifier (TWTA) transmission power noise higher than expected	TWTA transmission power noise ~1dB earlier in the mission, now reduced to ~0.2 dB or less. This noise has no effect on the quality of the data		Detector		None	
TOPEX microwave radiometer (TMR)- measures total water vapor along the path, as viewed by the altimeter, to correct altimeter data for pulse delay due to water vapor (accuracy 0.2-g/cm^2, equivalent to 1.2 cm).	1							

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effect On Individual Investigations:
PAYLOAD								
Laser retroreflector array (LRA)- used with ground-based lasers to track the satellite and to verify height measurement (accuracy 2 cm overhead ranging)	1 (no detailed info)							
DORIS dual Doppler tracking system receiver (DORIS)- receives signal from ground stations for satellite tracking (accuracy 5-10 cm altitude)	1[O]	First synchronization unsuccessful	Bad satellite time correlation table during command generation		Instrument	Used new synchronization sequence	None	
Single-frequency solid-state radar altimeter (SSALT)- measures height to precision goal of 5 cm altitude.	1[O] 2[M]	The first turn-on of the instrument unsuccessful. Mispointing of instrument angle	Procedural problem Initially 0.2¡-0.6¡, improved to 0.2¡, but still outside the requirement of 0.08¡ (1s).		Instrument Instrument	Performing a new turn-on sequence. Software patch	None Minor	
GPS demonstration receiver (GPSDR) - provides a new tracking data type (range differences) for continuous precision orbit determination (accuracy goal 10 cm or better altitude)	1							

OBJECTIVE	MET? (total, partial, not)	COMMENTS
LIFETIME:		
The mission is designed for a 3 year life, with enough margin on expendable resources for extension to 5 years.	not met	Loss of contact with S/C after initiation of propulsion system pressurization.
% OF LIFETIME MET	~30	
SCIENCE OBJECTIVES:		
Determine the global elemental and mineralogical character of the surface material.	not met	Loss of contact with S/C after initiation of propulsion system pressurization.
Define globally the topography and gravitational field.	not met	Loss of contact with S/C after initiation of propulsion system pressurization.
Establish the nature of the magnetic field.	not met	Loss of contact with S/C after initiation of propulsion system pressurization.
Determine the time and space distribution, abundance, sources, and sinks of volatile material and dust over a seasonal cycle.	not met	Loss of contact with S/C after initiation of propulsion system pressurization.
Explore the structure and aspects of the circulation of the atmosphere.	not met	Loss of contact with S/C after initiation of propulsion system pressurization.

CRUISE

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effec Indiv Inves
TELECOM								
Except during known or specified time intervals, provide a two-way communications link between the spacecraft and the DSN.	1							
For commands, provide for an uplink at any specified data rate, at X-band.	1							
Accommodate a Ka-band link experiment - not mission critical.	1							
Provide HGA and LGA capability	2[O]	Reduction in performance, not critical (during cruise only)	HGA misaligned 1i causing 6dB loss of signal in ANS- reduction in down-link rate		Data rate	Margin allowed, nominal performance	none (in cruise only)	None
COMMAND AND DATA								
Decode, store, execute and distribute commands	1							
Provide S/C clock.	1							
Acquire and format all engineering and science data	1							
Record and play back data at command times and rates.	1							
ATTITUDE AND ARTICULATION								
Provide 3-axis S/C stabilization, orientation and pointing.	2 [H]	temporary loss of reference	Solar proton activity caused anomaly		inertial reference	self healing	none	none
Sun Sensor Assembly	2 [M]	performance slightly degraded but performance was nominal.	Misaligned sun sensor head	ISA 4571	Control error in sun coning	Multiple heads	none	none
	2 [R]	Switch to backup sensor	Sun Sensor Assy Head in shadow of solar array	ISAs 3903 3920	Sun pointing	Redundant sensor	none	none

Celestial Sensor	2 [R]	Temporary loss of one fan	Caused by solar protons pulsing slit	ISA 4556	Fan shutdown	Redundant fans	none	none
Control major and minor propulsive maneuvers.	1							
Articulate 2 DOF solar panel to platform within accuracy.	1							

CRUISE

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effect On Individual Investigations:
PROPULSION/PYRO								
Provide thrust for payload insertion into interplanetary orbit by Titan III launch vehicle and TOS (transfer orbit stage)	1							
Provide thrust for trajectory control maneuvers. BPE (Bipropellant Equip) 4x490 N bipropellant rocket engine (BREs) and 4x22N bipropellant rocket engines MPE (Monopropellant Equip) 8x4.45 N catalytic rocket engine assemblies (REAs) and 4x0.9 N catalytic REAs BPE + MPE used for TOS separation, TCM 1 through 4 (3 actual performed), Mars capture and transfer to observational orbit & boost to quarantine orbit. In addition, MPE used for Mars orbit trim maneuvers.	2 [R]	Temp. specs. violated	Failure of thermostat	ISA 3794	NTO tank cooled too rapidly	Redundant temp control	no	no
Provide thrust for attitude control maneuvers.	1							
Provide for the firing of pyro squibs.	1							

Initiation of Propulsion System Pressurization Prior to MOI

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effect On Individual Investigations:
PROPULSION/PYRO								
Provide for insertion into Mars orbit:								
BPE	5[A]	Catastrophic loss of control of S/C	Unintended mixing of NTO/ MMH causing a breach of the pressurization plumbing	*	All subsequent events and functions	none	catastrophic	catastrophic
	5[B]	Catastrophic loss of control of S/C	Regulator failed open /tank over pressure causing a rapture	**	All subsequent events and functions	none	catastrophic	catastrophic
	5[B]	Catastrophic loss of control of S/C	Pyro valve failure, NSI expelled causing NSI to puncture critical S/C component(s)	**	All subsequent events and functions	none	catastrophic	catastrophic

*5A = most probable cause of MO failure - (Ref. 15 and 16)
**5B = potential cause of MO failure - (Ref. 15 and 16)

CRUISE

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effect On Individual Investigations:
POWER								
S/C shall provide power via power supply SS prior to launch until end of mission.	1							
Primary power shall be supplied via a photovoltaic solar array.	2 [M]	Array operation at higher temp. - no perf. impact	solar array operated hotter than predicted	ISA 3862	Array power output	Ample margin	minor	none
Provide for power control, regulation & distribution.	1							
Provide for firing of squibs.	1							
Provide for energy storage, recharged batteries.	1							
Provide telemetry signals for monitoring & control of EPS.	1							

Initiation of Propulsion System Pressurization Prior to MOI

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effect On Individual Investigations:
POWER SUBSYSTEM								
Protect itself from short circuits and shorts to S/C structure.	5[B]	Shut down of power distribution to S/C	Power supply electronics power diode insulation failure resulting in a short between power bus and chassis.	** (See propulsion system notes)	S/C power distribution; and all subsequent events and functions	none	Catastrophic	Catastrophic

**5B = potential cause of MO failure - (Ref. 15 and 16)

CRUISE

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effect On Individual Investigations:
THERMAL								
Maintain control of the temperature of all S/C engineering and science subsystems and their elements/components within specified limits. Aid in providing micro meteorite protection.	2 [O]	Temp-monitoring of MOC narrow angle electronics erratic	failed thermister resulted in erratic temp data	ISA 3772	temp monitoring	alternate telemetry available	none	none
STRUCTURE								
Provide mechanical support and alignment; provide radiation and micrometeroid protection; protect equipment from excessive vibration, etc; aid in assuring an equipotential surface.	1							
CABLING								
Provide the electrical interconnections between all S/C engineering and science subsystems, between subsystem elements, and those needed for interfacing with a separable vehicle system, with the L/V, with jettisonable elements, and with the launch complex. Provide for appropriate shielding, EMI protection, and grounding	1							
MECHANISMS								
Solar panel deployment assembly.	1							
Solar array gimbal drive assembly.	1							
High gain antenna deployment assembly including HGA gimbal assembly.	2 [M] (Function performed late but in adequate time)	Full deployment of high-gain antenna	Mechanism deployed HGA late, possibly due to cold stiff cable or bolt head hang-up	ISA 3912	HGA	deployment none	none	none

Gamma ray spectrometer (GRS) deployment assembly	1							
Magnetometer/Electron reflectometer (MAG/ER) deployment assembly.	1							

Engineering Performance Criteria	*Perf. Met?	Performance Deviation	Anomalies Affecting Perf.	Date (PFR #)	Function Affected	Fix Description	Mission Impact	Effect On Individual Investigations:
Payload								
Mars Observer Camera	2 [O]	Distortions in images of narrow angle optics	Astigmatism of the narrow angle optics		Image quality	Application of about 1 watt heat to mirror rim heater and operating the camera at the point of best focus	none	none
Gamma Ray Spectrometer	****							
Pressure Modulator Infrared Radiometer	2 [O]	Operating below required temp	Auxiliary power turned off during maneuver	ISA 3914	Normal operations	Permanently leave auxiliary power on	none	none
Mars Observer Laser Altimeter	****							
Thermal Emission Spectrometer	****							
Magnetometer/ Election Reflectometer	4 [O]	Magnetic interference from S/C	Large magnetic fluctuations caused by S/C current loops resulting from solar array operation	ISA 4578	Ability to map Mars dynamic and static magnetic field components	Map magnetic field during solar occultation would allow substantial data to be obtained. If Martian magnetic field values at upper limit, it might have been possible to map at other times also.	minor	minor-significant
Electron Reflectometer	3 [M]	loss of lowest energy-range electrons	ER case shorted to S/C chassis grove	PFR 59014 ISA 4573	Ability to measure local surface magnetic fields component	none	Minor (ER data was supplemental data)	significant
Radio Science	1 ***							

Mars Balloon Relay (collaboration with the French/Russians, not a primary instrument)	****							

*** Gravity wave experiment successful

**** Instrument to be operated in Mars Orbit